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OFFICE OF MANNED SPACE FLIGHT

APOLLO PROGRAM

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APOLLO-SATURN 201 MISSION RELIABILITY ANALYSIS (U)

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APOLLO RELIABILITY AND QUALITY ASSURANCE OFFICE

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546

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APOLLO PROGRAM

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APOLLO-SATURN 201

MISSION RELIABILITY ANALYSIS (U)

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SECTION 1

AS-201 MISSION ANALYSIS REPORT

1.1 INTRODUCTION

This analysis was conducted to predict the reliability of the Apollo-Saturn 201 mission; an unmanned flight with a suborbital, lob-type trajectory along the Eastern Test Range, with impact in the vicinity of Ascension Island.

This will be the first flight of the S-IB stage using H-1 engines upgraded in thrust, of the S-IVB stage with a single J-2 engine, and of a production Block 1 Apollo spacecraft (except for deletion of the guidance and navigation package and certain subsystems associated with crew requirements, and use of batteries instead of fuel cells).

The principal mission objectives are to obtain information about actual flight performance of the stages, structural integrity and compatibility of launch vehicle and spacecraft, and performance of the command module heat shield during re-entry at maximum heat rate. The flight duration is approximately 40 minutes as compared to 14 days for a nominal lunar landing mission, and there is no S-II stage. Consequently, operation of the S-IB stage is quite similar to that required for later missions, the requirements for the S-IVB stage are somewhat different, and those for the spacecraft are much less severe. Thus, since the equipment to be used for the AS-201 mission has been designed for the requirements of lunar missions, it should be satisfactory for these less severe conditions.

Using AS-201 data supplied by the MSF Centers, the launch vehicle contractors, and supplemental data from the AS-204 and AS-504 mission analyses performed by the spacecraft contractors, a reliability profile and model explicitly representing the AS-201 mission was generated and a reliability analysis was conducted. The effects of ground support equipment (GSE), the ground operational support system (GOSS), and prelaunch operations (launch availability) have been omitted because of insufficient reliability data.

The predicted reliability for the AS-201 mission exceeds the NASA goals. Since succeeding missions will place increasing demands on the equipment, this fact does not indicate that succeeding missions will continue to show a comfortable reliability margin. The results of this AS-201 analysis do not necessarily indicate that succeeding missions using similar equipment will have correspondingly high reliabilities.

To obtain an over-all perspective on the reliability status of the AS-201 mission, the pros and cons of several measures are tabulated with a summarizing conclusion for each. These follow:

Measure	Pro	Con
Reliability Analysis	1. Contractors' predicted reliability based on historical data meets requirements. 2. Spacecraft equipment designed for long life but AS-201 mission only 40 minutes duration. 3. All analyses conducted indicate capability to exceed goals. 4. Previous Saturn successes tend to provide assurance.	1. Analyses specifically for AS-201 were necessarily late and insufficiently comprehensive. 2. Much of equipment being flown for first time.

CONCLUSION:

Although there are insufficient analyses for complete assurance of success, the analyses conducted have not disclosed any evidence of design inadequacies.

Measure	Pro	Con
Test Results	1. Contractors' reliability assessments that have been conducted confirm predictions.	1. Reliability assessments are incomplete and based on assumed successful tests.
	2. Preflight checkouts will confirm readiness.	2. Criteria, such as number of required successful checks, are not known to exist.
	3. Ground firings successful.	3. Only limited systems tests have been conducted.

CONCLUSION:

Because of limitations in testing accomplished, checkout criteria should be particularly stringent.

Measure	Pro	Con
Qualification Status	1. Contractors and Centers are making concerted effort to certify qualification.	1. Not all equipment is qualified.
	2. Qualification requirements are stricter than those required for AS-201.	2. Some rationalizations are categorized "questionable."

CONCLUSION:

Expected verification of rationalizations will satisfy AS-201 qualification requirements.

Measure	Pro	Con
Open Items	1. No insurmountable problems have been disclosed. 2. Contractors and Centers are making conscientious effort to correct problems. 3. Corrective action is being taken on reported failures.	1. Total unresolved problems are not decreasing.

CONCLUSION:

The number of unresolved problems is a matter of serious concern.

Measure	Pro	Con
Criticality Determinations	1. Principal contributors to unreliability have been identified. 2. Special emphasis is being given to critical areas. 3. Action is being taken to improve spares provisioning. 4. No apparent reliability problems have been identified which prevent completion of prelaunch.	1. No evidence that all single failures have been identified. 2. Spares are generally lacking. 3. Potential problems exist in verification of operational procedures.

CONCLUSION:

The major risk items have been identified as the S-IB propulsion and mechanical subsystems, the S-IVB propulsion and mechanical subsystems, the IU guidance and control subsystems, and the S-IVB flight control subsystem. (See Table 2-4 for details.)

Measure	Pro	Con
Launch Complex	1. LC 34 modification and checkout is complete. 2. Delineated problem areas receiving high priority attention. 3. Launch Area Readiness Assessment Report ready 29 December 1965.	1. First launch after extensive modification of LC 34.

CONCLUSION:

Status of Operations and Checkout Building, Control Instrumentation Facility, Central Telephone Office, and Eastern Test Range relative to AS-201 must be determined.

Measure	Pro	Con
Interfaces	1. Checkout procedures and mock countdowns provide integrated systems evaluation. 2. Predictions for probable achievement of mission objectives evaluate top level interdependencies. 3. Simplicity of mission unmanned lob shot reduces interface requirements.	

CONCLUSION:

Apparently no unresolved interface problems exist.

Reliability mission analyses can be of great assistance in determining optimum plans, in terms of crew safety or mission success, for use by the Apollo Flight Director. The choice of the optimum abort mode or alternate mission can also be aided by use of this analysis. Because of the simplicity of the AS-201 mission, as compared to the manned lunar landing mission, there was no need or opportunity to establish elaborate mission ground rules.

For certain types of missions these analyses can also be used to determine optimum deployment of recovery forces by determining the probabilities of landing in certain specified landing areas.

On the AS-201 mission, this report makes clear that the bulk of the reliability problems lie with the launch vehicle rather than the spacecraft. While the nature of this mission makes this conclusion somewhat obvious, if this same relationship continues for the oncoming missions, a further review of allocation of resources may be required.

Reliability assessments and predictions, as used in this report, are defined by NPC 250-1 (Reference 16) as:

> RELIABILITY ASSESSMENT. An analytical determination of numerical reliability of a system or portion thereof. Such assessments usually employ mathematical modeling, use of directly applicable results of tests on system hardware, and some use of estimated reliability figures.
>
> RELIABILITY PREDICTION. An analytical prediction of numerical reliability of a system or portion thereof similar to a reliability assessment except that the prediction is normally made in the earlier design stages where very little directly applicable test data is available.

SECTION 2

APOLLO-SATURN 201 SUMMARY

2.1 GENERAL

A systematic reliability analysis of the Apollo-Saturn 201 mission was conducted. The reliability of the over-all mission was calculated using Center and contractor inputs where available (launch vehicle) supplemented by models and data generated by the Apollo Reliability and Quality Assurance Office (spacecraft). A model of the over-all reliability requirements was prepared based on AS-201 mission profile. (Applicable information from the AS-504 and AS-204 modeling activities was used where AS-201 data was not available.) This analysis contains the first combined launch vehicle and spacecraft reliability forecasts for the AS-201 flight vehicle. No major change in these forcasts is expected.

The "all engines required" data was used for the S-IB since agreement has not been reached on "one engine out" capability data for this mission.

2.2 RESULTS

The following results were achieved:

a. The reliability predictions for success of the Apollo-Saturn 201 mission exceed program specification goals.
b. Based on these predictions, the odds for successfully achieving the Apollo-Saturn 201 mission objectives are 14 to 1.
c. The predicted values for each of the flight stages exceed the requirements given in the <u>Apollo Program specification</u> (Table 2-1).

The best available reliability prediction values were obtained for similar equipment from other programs and modified, when necessary, for specific Apollo-Saturn usage. These prediction values were replaced by actual Apollo-Saturn test data in 37 flight critical items in the S-IVB Stage. Similarly, flight critical items of the S-IB stage and instrument unit were reviewed, but the available test data was considered insufficient, by the stage contractors, for the making of significant changes in the predicted reliabilities. This limited substitution resulted in little change in the final results (see last column of Table 2-1).

Table 2-1

Comparison of Apollo-Saturn 201 Predictions to Program Specification Goals

Apollo Program Specification (Reference 2) (Its Appendix AS-201, paragraph 3.1.3.3)			Mission Success Prediction	Reference for Prediction	Preliminary Assessment ****
Phase	Equipment	Success Probability Goal			
Flight*	S-IB	0.95	0.99 (or 0.95)**	(18 or 19)	0.96
	S-IVB	0.95	0.97	21	0.96
	IU	0.99	0.99	27	0.99
	CSM/LES	0.96	0.98	(Appendix A, This Report)	
	LEM/Adapter	0.98	1.0***		
	Complete Flight Assembly	0.85	0.93 (or 0.90)	(This Report)	0.90

* The flight phase begins with space vehicle from the launch pad and terminates with recovery of the CM.

** The S-IB-1 contractor furnished two predictions, one based on Failure Mode Effects and Criticality Analyses, the other based on 10,000 simulated flights.

*** Adapter structure only. There is no LEM in AS-201 (adapter separation is included in CSM/LEM number).

**** Based on inclusion of test data available December 1965.

The scarcity of reliability test information emphasizes the necessity for a thorough review of the completeness and adequacy of the test program and quality control practices employed in preparing the equipment to meet the Apollo-Saturn 201 mission objectives, since these activities assure that the inherent reliability of the equipment has been maintained throughout manufacturing and field operation.

2.3 CONCLUSIONS

The following conclusions are drawn:

a. The mission success predictions for the Apollo-Saturn 201 mission exceed the program specification goals. Based on limited Apollo-Saturn test data and best reliability prediction data available at this time, the design of the flight equipment is satisfactory for the purposes of this mission.

b. The results of this analysis can be used as a "baseline" from which to assess the effects of unresolved quality or test problems encountered with the equipment to be used for the Apollo-Saturn 201 mission. Descriptions of such problems and the corrective action taken should be furnished by the Centers and their contractors prior to the flight readiness reviews.

c. Relatively high unreliability of the H-1 and J-2 engines were reported by the stage contractors. NAA/Rocketdyne is performing a reliability analysis of the engines which could alter the results described in this report.

d. The major reliability problems in the AS-201 mission are being reviewed by MSC, MSFC, and their contractors. Information on courses of action being undertaken and those proposed to be taken to solve the reliability problems, the tests being performed and those proposed to be performed, etc., are being gathered presently by the contractors.

e. The profile and environmental requirements of the AS-201 mission are established, and the flight equipment design is suitable for the intended use. The unevaluated or potentially controversial areas remaining are the checkout and test status of the flight equipment (unsatisfactory conditions, etc.), the adequacy of the planned telemetry and tracking (Eastern Test Range), and the suitability and adequacy of the Launch Complex 34 equipment and procedures.

2.4 OBJECTIVE REALIZATION

The six primary objectives for the Apollo-Saturn 201 mission, as listed in the OMSF Flight Mission Directive (Reference 1), cover the demonstration, verification, or evaluation of 23 functions or subsystems. For purposes of analysis and review, the objectives were paraphrased to list these 23 items individually.

The over-all probability (including the effects of all phases of the flight mission required through the completion of each objective) of meeting each of these detailed objectives was determined and the results are presented by dots in Table 2-2. For applicable objectives, the conditional probability of meeting the objective (provided the mission has proceeded to the point where accomplishment of the particular objective has been started) was also obtained and is shown for the purpose of comparison by deltas (▲) in the same illustration. The conditional probabilities of success will always be higher than the corresponding over-all probabilities. Risk factors associated with the primary objectives, computed from the paraphrased objective results, are shown in Table 2-3.

The odds of meeting parts of the mission objective parts vary from a high of 99 to 1 (corresponding to the success probability of almost 0.99) for determination of the structural loading of the spacecraft adapter when subjected to the launch environment, to a low of 14 to 1 (or a success probability of a little over 0.93) for determination of the structural integrity of the spacecraft. Such odds indicate the uncertainty remaining after the effects of known problem areas have been minimized by the best efforts of all program personnel.

2.5 MISSION ANALYSIS

For this analysis, the Apollo-Saturn 201 mission was divided into phases that can be readily monitored during the flight, and reliability predictions were made for each phase. The over-all probability of successfully completing each phase of the mission is shown in Figure 2-1. Each phase extends from an event to the next event.

Starting with an assumed reliability of 1.0 at holddown release, the cumulative odds of successfully completing each selected portion of the mission profile vary from a high of 66 to 1 for completion of the S-IB engine burn to a low of 14 to 1 for completion of retrieval after splashdown, with the principal degradation occurring in phases 2 and 5 when the launch vehicle propulsion systems are in operation.

The calculated over-all probabilities of mission success are plotted against a linear time scale in Figure 2-2 for the flight interval from liftoff to touchdown.

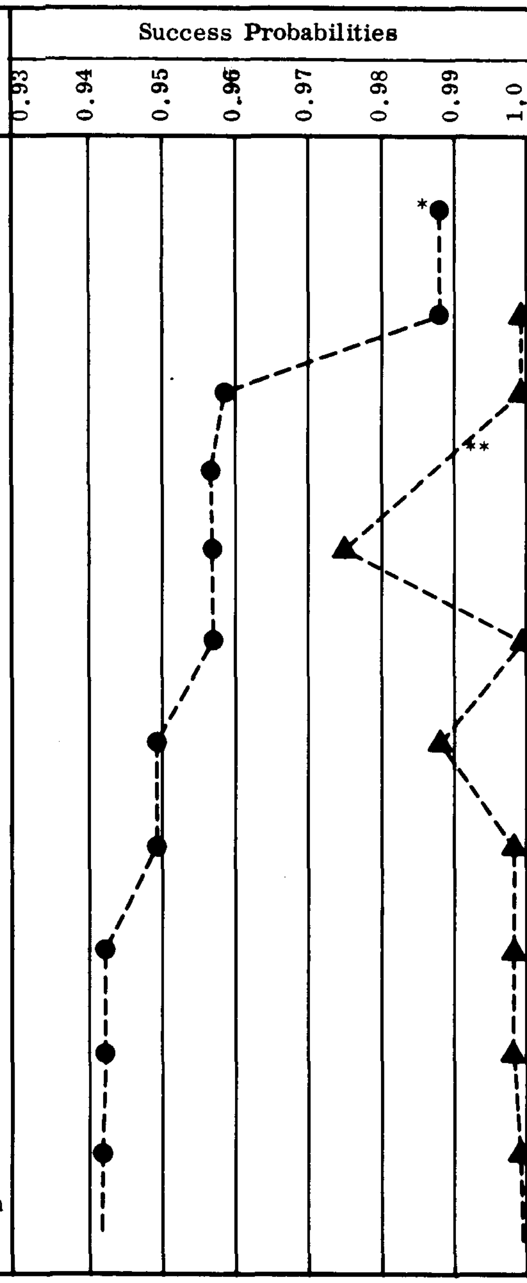
The reliability predictions for mission success, as calculated for the over-all Apollo-Saturn 201 flight, were converted to unreliability values for each vehicle stage. The relative effect of each stage on the predicted unreliability of the mission is shown in Figure 2-3. The launch vehicle stages account for over one-half of the over-all unreliability.

2.6 PRINCIPAL CONTRIBUTORS TO UNRELIABILITY

The subsystems having the largest effects on unreliability, and the mission phases during which these contributions occur, were identified. The contributions to unreliability are summarized as percentages in Table 2-4. The references (Appendix C in this report) from which the prediction data was obtained are also shown. The subsystems and phases listed account for over 90 percent of the total unreliability of the AS-201 mission.

Table 2-2

Apollo-Saturn 201 Mission Objective - Predictions



Mission Objectives (Paraphrased)	Success Probabilities	Risk*** Factor (Percent)
1. Determine structural loading of the spacecraft adapter (SLA) when subjected to the Saturn IB launch environment.		1
2. Demonstration of S-IB S-IVB separation.		1
3. Demonstration of launch escape system separation.		4
		4
4. Demonstration of launch vehicle structural integrity.		
5. Verification of launch vehicle propulsion subsystem operation.		4
6. **Evaluation of performance of the open-loop emergency detection subsystem.**		4
7. Verification of launch vehicle guidance and control subsystem operation.		5
8. Demonstration of launch vehicle - command/service module (CSM) separation.		5
9. Determine long-duration service propulsion subsystem (SPS) performance including shutdown.		6
10. Demonstrate restart of service propulsion subsystem (SPS) following long-duration burn.		6
11. Verification of spacecraft service module reaction control subsystem (SM-RCS) operation.		6

* Over-all, including the effects of all phases of the flight mission required through the completion of each objective.

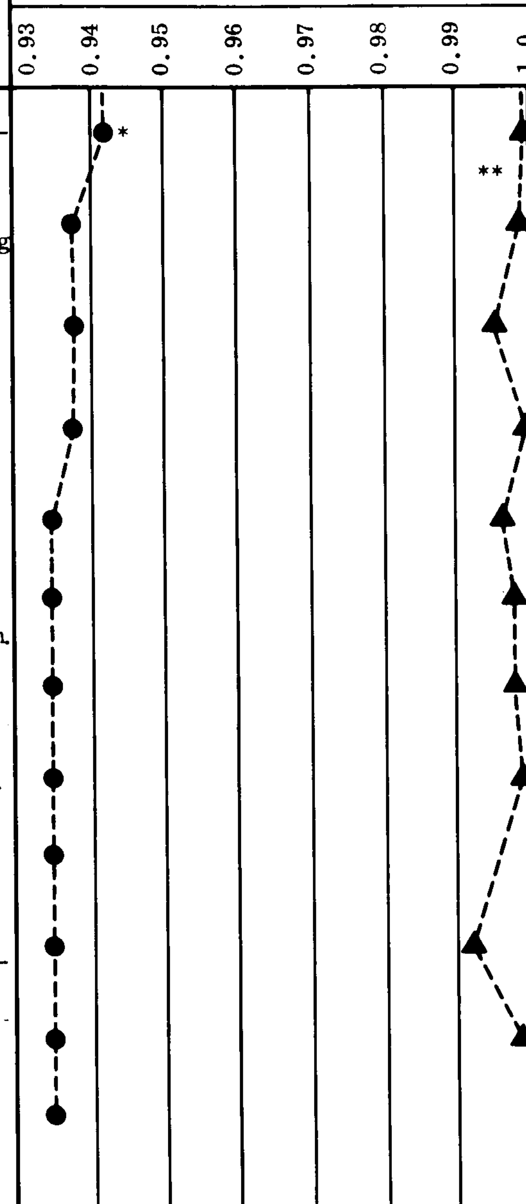
** Conditional, providing the mission has proceeded to the point where accomplishment of the particular objective has been started.

***Risk factor is defined as one minus the probability of success, expressed as percent.

Table 2-2

Apollo-Saturn 201 Mission Objective - Predictions (Continued)

Mission Objectives (Paraphrased)	Success Probabilities	Risk*** Factor (Percent)
12. Demonstration of service module-command module separation.		6
13. Evaluate command module heat shield ablator performance during high heat-rate entry.		6
14. Verification of spacecraft stabilization control subsystem (SCS) operation.		6
15. Verification of spacecraft command module reaction control subsystem (CM-RCS) operation.		6
16. Verification of spacecraft communication subsystem operation.		7
17. Verification of spacecraft earth landing subsystem (ELS) operation.		7
18. Verification of spacecraft environmental control subsystem (ECS) operation.		7
19. Verification of spacecraft electrical power subsystem (EPS) operation.		7
20. Determine adequacy of recovery aids.		7
21. Determine command module adequacy for entry from low earth orbit.		7
22. Demonstration of spacecraft structural integrity.		7
23. Demonstration of mission support facilities.		7



* Over-all, including the effects of all phases of the flight mission required through the completion of each objective.

** Conditional, providing the mission has proceeded to the point where accomplishment of the particular objective has been started.

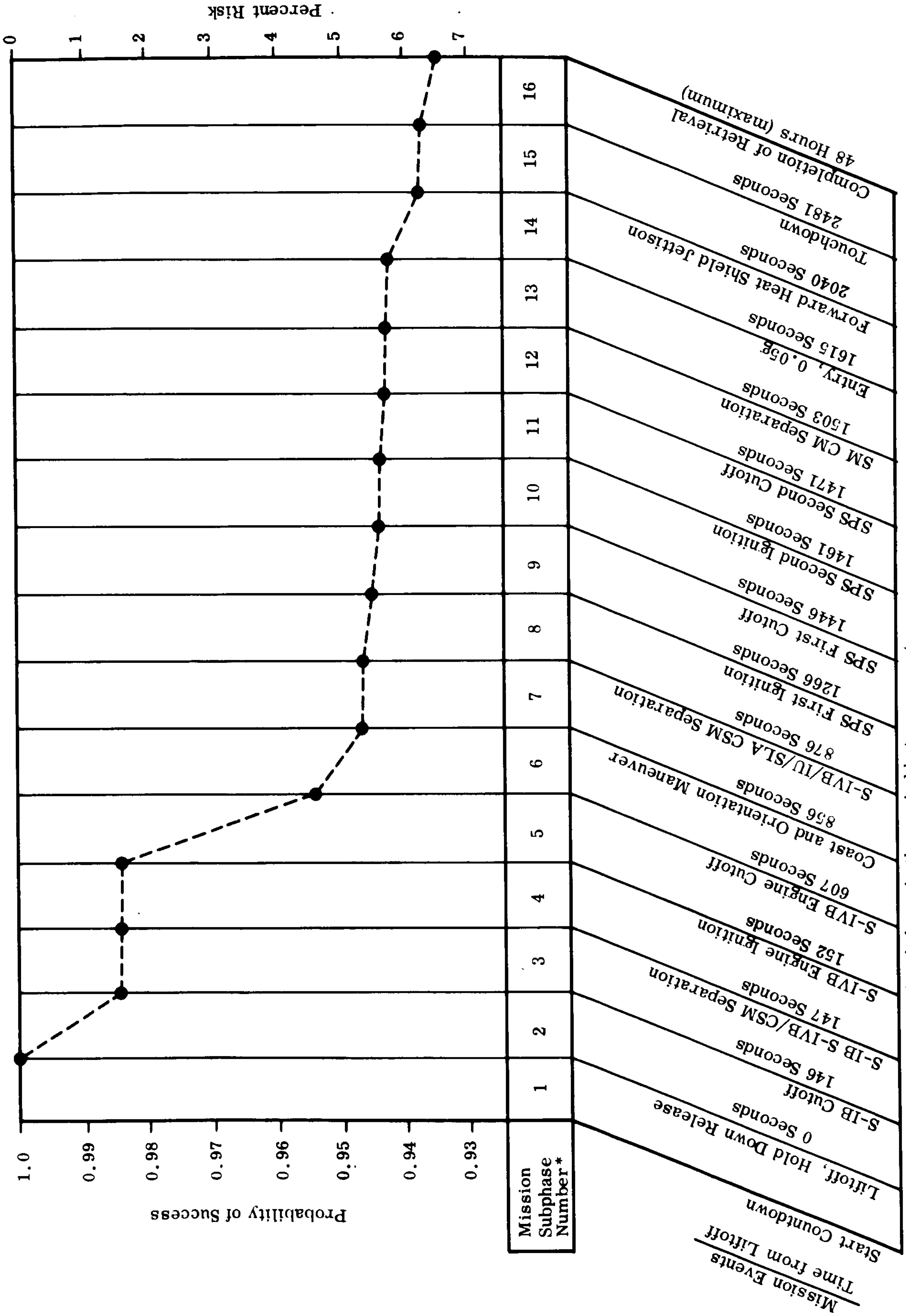
***Risk factor is defined as one minus the probability of success, expressed as percent.

Table 2-3

Predictions for Primary Objectives

Primary Objectives of the Apollo-Saturn 201 Mission	Contributors* (Paraphrased Objective Numbers)	Prediction for Objective Success		Major Contributors to Risk
		Success Probability	Risk Factor %	
(A) Demonstration of structural integrity and compatibility of the launch vehicle and spacecraft and determination of launch loads.	1, 4, 22	0.93	7	S-IB Propulsion and Mechanical Subsystem
(B) Demonstration of separation of S-IVB from S-IB, LES, and Boost Protective Cover from CSM, CSM from S-IVB/IU/SLA, and CM from SM.	2, 3, 8, 12	0.94	6	S-IVB Electrical Subsystem S-IVB Propulsion and Mechanical Subsystem
(C) Verification of operation of the following launch vehicle and spacecraft subsystems.				
(1) Launch Vehicle: propulsion, guidance, and control.	5, 7	0.95	5	S-IVB Flight Control Subsystem
(2) Spacecraft: CM heat shield (adequacy for entry from low earth orbit) SPS (including restart): ECS (pressure and temperature control); communication (partial); CM RCS; SM RCS; SCS; ELS; EPS (partial).	9, 10, 11, 14 15, 16, 17, 18, 19, 21	0.93	7	CSM Stabilization and Control Subsystem CSM Service Propulsion Sybtem
(D) Evaluation of open-loop space vehicle Emergency Detection System.	6	0.96	4	IU Guidance and Control Subsystem
(E) Evaluation of the CM heat shield at a high heat rate of approximately 200 Btu/ft^2-sec during entry at approximately 28,000 ft/sec.	13	0.94	6	CSM Communication Subsystem
(F) Demonstration of the mission support facilities required for launch, mission operations, and CM recovery.	20, 23	0.93	7	Incomplete Assessment No Contributors Identified

*Each OMSF Primary Objective will be achieved when the last of its contributing "paraphrased objectives" is completed; hence, the probability of completing the composite objective is the same as that for completing the contributing objective which is reached at the latest time during the flight.



*A subphase is the period between events.

Figure 2-1. Apollo-Saturn 201 Mission Success Predictions Graph

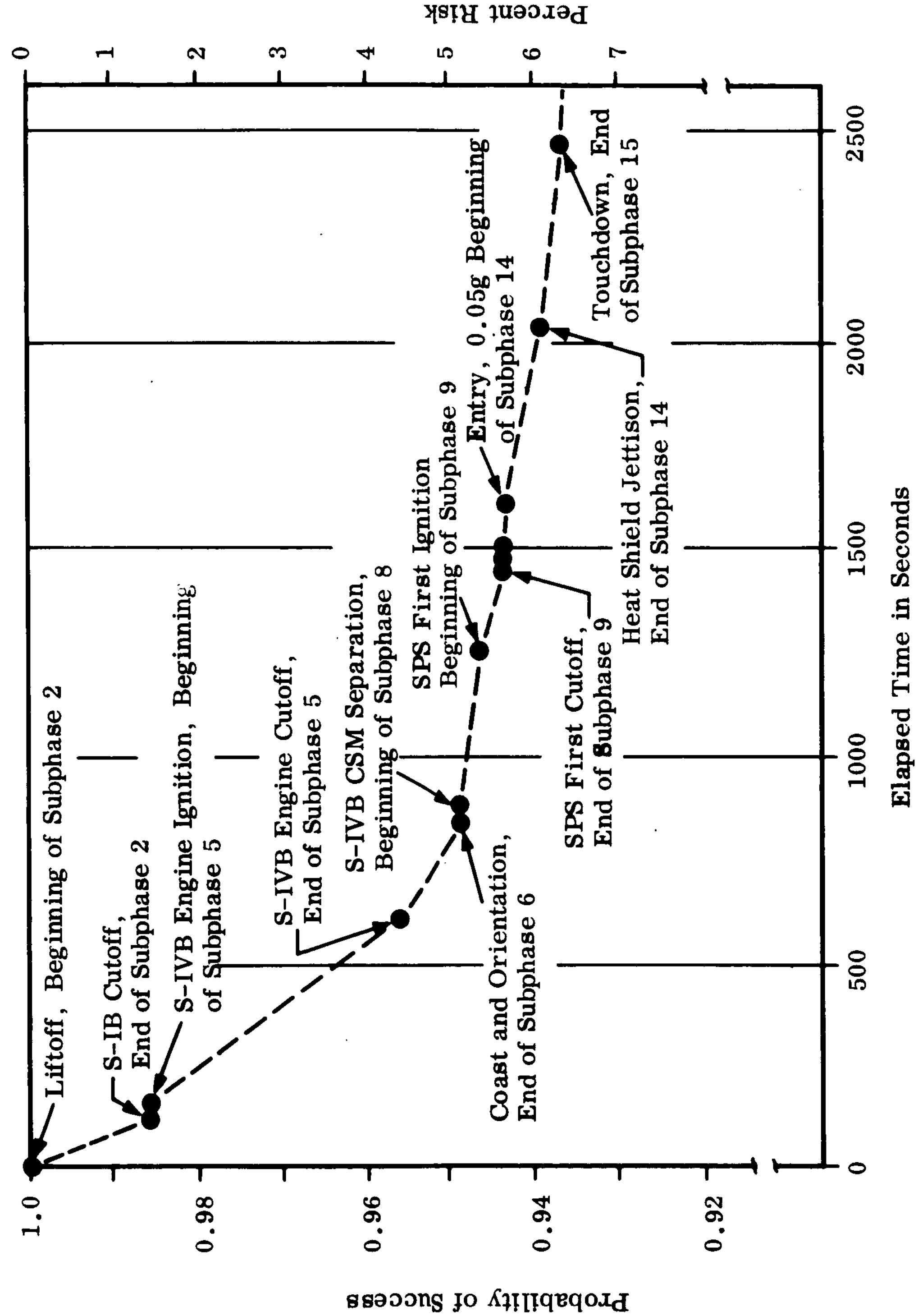


Figure 2-2. Apollo-Saturn 201 Mission Success Predictions

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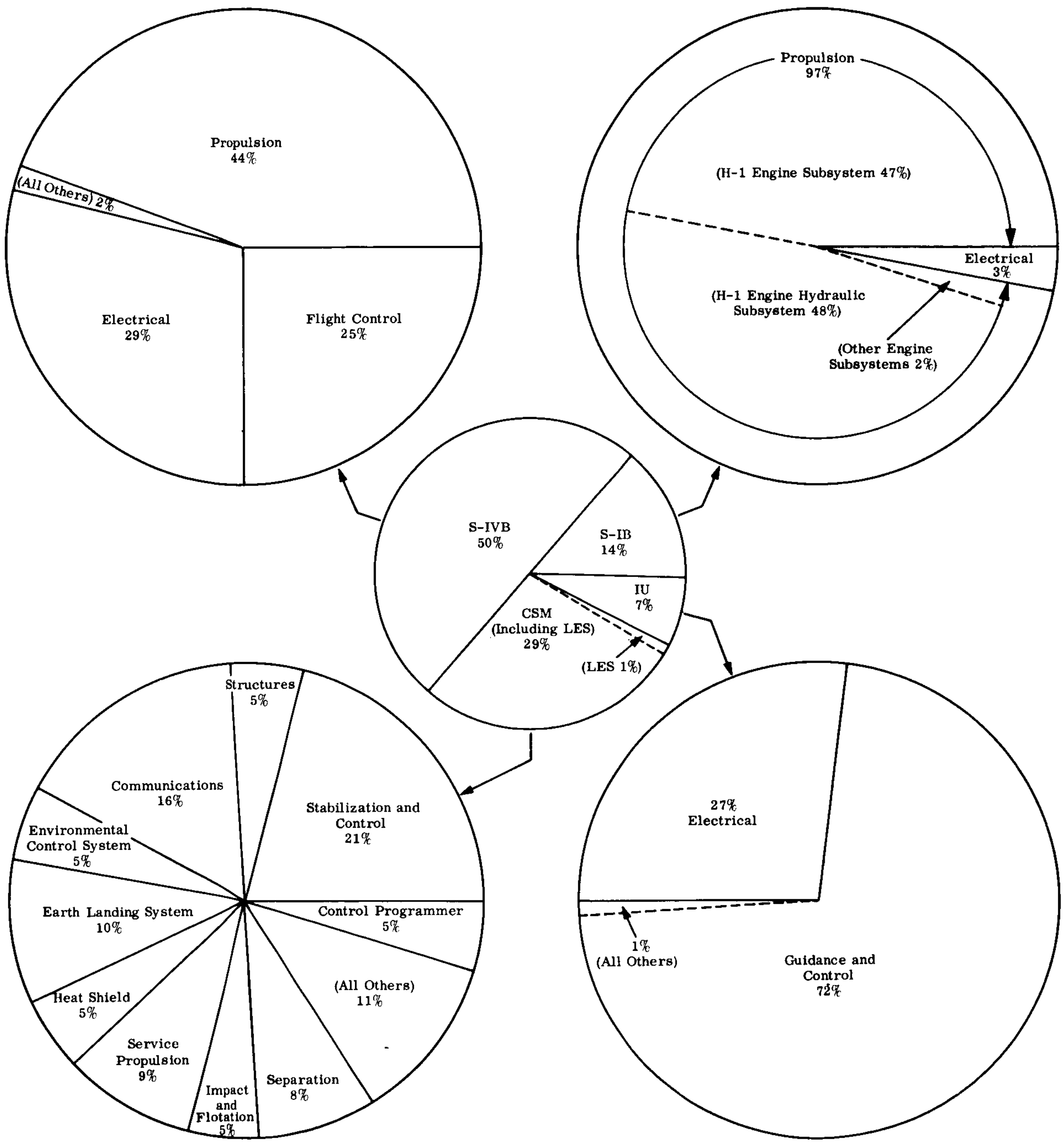

Figure 2-3. Apollo-Saturn 201 Mission, Percent Contribution to Unreliability Based on Predictions

[redacted]

[redacted]

Table 2-4

Main Contributors to Unreliability

Events (A phase extends from an event to the next event)	Mission Phase: Phase Number	Mission Phase: Percent Contribution of Phase to Unreliability of Mission	Principal Contributors: Subsystem	Principal Contributors: Percent Contribution of Subsystem of Unreliability of Phase	Reference (Source of Subsystem Data)
Liftoff, Hold Down Release	2	22	S-IB Propulsion	63	18
			S-IVB Electrical	9	21
			IU Giudance and Control	6	27
S-IB Cutoff			S-IVB Flight Control	6	21
			IU Electrical Power	4	27
S-IVB Engine Ignition	5	46	S-IVB Propulsion	47	21
			S-IVB Electrical	19	21
			IU Guidance and Control	8	27
S-IVB Engine Cutoff			S-IVB Flight Control	8	21
			CSM Stabilization and Control	5	26, 31
S-IVB Engine Cutoff	6	11	S-IVB Flight Control	67	21
Coast and Orientation, End			S-IVB Electrical	29	21
Entry, 0.05g	14	6	CM Stabilization and Control	36	26, 31
			CM Communications	27	20, 26
Heat Shield Jettison			CM Heat Shield	25	33
			CM Instrumentation	6	20, 26
Touchdown	16	5	CM Earth Landing System	57	29
			CM Impact and Flotation	28	34
Retrieval			CM Electrical Power	12	26, 30, 33

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The two major areas of unreliability are associated with the propulsion subsystems of the S-IB and S-IVB. The identification of these propulsion subsystems as major contributors does not imply the existence of design deficiencies or unresolved problem areas; rather it points out the importance and state of development of these major items (see paragraph 2.3, subparagraphs d and e). The ground testing of both the S-IB and S-IVB propulsion systems has been extensive. The AS-201 flight test is by plan the first test of these two particular configurations. Since all the combinations of flight environmental parameters cannot be duplicated during tests conducted on the ground, some uncertainty will remain when the equipment is subjected to its first flight test. From the standpoint of these propulsion subsystems, the AS-201 flight test can be considered as a planned step in the continuing development of these equipments.

Since the over-all predicted unreliability for the AS-201 mission represents a calculated risk which is acceptable according to the Apollo Program Specification (Reference 2) goals, the contribution of each of these subsystems is also considered as an acceptable calculated risk.

Since the AS-201 flight is a preparatory mission, or a planned step in the over-all Apollo development plans, the data which will be obtained during this operation will contribute to greater knowledge of the equipment capabilities and hence to the minimizing of the unreliabilities of subsequent launches.

2.7 MAJOR MALFUNCTION CONTINGENCIES

In the event of major malfunction of the 201 launch vehicle, action may be initiated to permit recovery of the command module. Provisions have been made in the mission plans for two types of contingencies (often called aborts). Power for command module separation can be provided by the launch escape system (LES) at any time from S-IB ignition to the jettisoning of the LES after S-IVB ignition, and by the service propulsion system (SPS) at any time from LES jettisoning to S-IVB CM separation. Either contingency must be initiated by command from the ground since the malfunction detection system (MDS) will be operated in an open-loop condition, with its operation monitored by telemetry. For either contingency the earth landing system (ELS) operates to control the descent of the command module to a soft landing.

The probability of contingency success will vary with the conditions existing at the time of initiation of the contingency. The worst case of each contingency was modeled.

These "worst cases" occur at the end of their respective contingency periods. If a contingency is required earlier in a period, its success probability will be higher than the associated worst case values.

If the launch escape system contingency is initiated, the odds according to predictions are 142 to 1 that it will be successfully completed. Similarly, the odds of the contingency using the service propulsion system being successfully completed are 90 to 1.

[redacted]

SECTION 3

APOLLO-SATURN 201 MISSION

3.1 GENERAL

The first stage path of Apollo-Saturn 201 will approximate that to be used later for orbital missions. The launch will be the first of a series using the S-IB stage with the H-1 engines upgraded in thrust, and the S-IVB stage with a single J-2 engine replacing the previously used cluster of RL-10 engines. The vehicle configuration is shown in Figure 3-1.

3.2 MISSION OBJECTIVES

The objectives for the Apollo-Saturn 201 mission are specified by the OMSF Flight Mission Directive (Reference 1). These objectives are quoted here as an extract from that document:

> Primary objectives of the Apollo-Saturn 201 Mission are tabulated below. Malfunction of spacecraft of launch vehicle systems, ground equipment, or instrumentation which would result in failure to achieve these objectives will be cause to hold or cancel the mission until the malfunction has been eliminated.
>
> a. Demonstration of structural integrity and compatibility of the launch vehicle and spacecraft and determination of launch loads.
>
> b. Demonstration of separation of S-IVB from S-IB, LES and Boost Protective Cover from CSM, CSM from S-IVB/IU/SLA and CM from SM.
>
> c. Verification of operation of the following launch vehicle and spacecraft subsystems.
>
> (1) Launch Vehicle: propulsion, guidance and control.
>
> (2) Spacecraft: CM heat shield (adequacy for entry from low earth orbit) SPS (including re-start): ECS (pressure and temperature control); communication (partial); CM RCS; SM RCS; SCS; ELS; EPS (partial).
>
> d. Evaluation of open-loop space vehicle Emergency Detection System.
>
> e. Evaluation of the CM heat shield at a high heat rate of approximately 200 BTU/ft-sec during entry at approximately 28,000 ft/sec.

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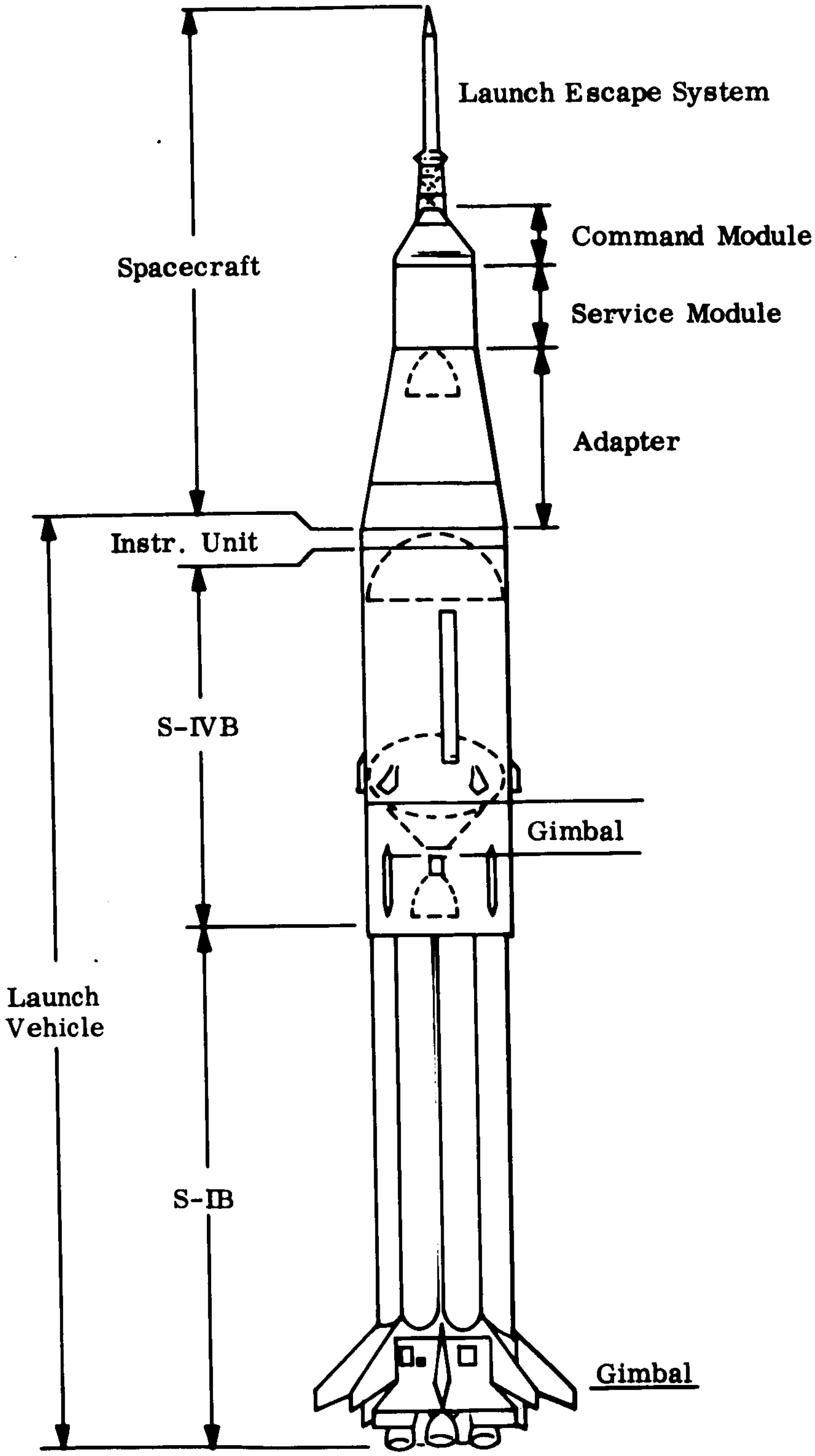


Figure 3-1. AS-201 Vehicle Configuration

[redacted]

f. Demonstration of the mission support facilities required for launch, mission operations and CM recovery.

3.3 MISSION FLIGHT PLAN

The Apollo-Saturn 201 mission calls for an unmanned flight following a ballistic trajectory along the Eastern Test Range. Launch will be from Complex 34 at Cape Kennedy, with touchdown in the vicinity of Ascension Island.

3.3.1 GROUND TRACK

The planned ground track for the Apollo-Saturn 201 mission is shown in Figure 3-2. This plot shows the coverage of the nine primary tracking stations and the approximate locations at which some of the mission events will occur. The recovery area is about 4800 nautical miles from the launch site.

3.3.2 TRAJECTORY

The planned trajectory for the Apollo-Saturn 201 mission is shown in Figure 3-3. This graph supplements Figure 3-2, and indicates the altitude at which some of the major mission events are planned. The maximum altitude reached will be a little over 270 nautical miles.

3.3.3 NORMALIZED PROFILE

The normalized profile for the Apollo-Saturn 201 mission was developed from detailed information contained in Saturn IB/SA-201 Flight Sequence (Reference 14) and Program Apollo Flight Mission Directive for Mission AS-201 (AFRM 009) (Reference 8). The events chosen for this profile were selected from the many detailed events listed in these center profiles, based on the subphase determination requirements.

For this analysis a phase is defined as an interval of time in the mission extending from a selected major event to another major event. The complete mission is then described by the sum of the modeled phases.

The simplified mission profile time line used in this analysis is shown in Table 3-1.

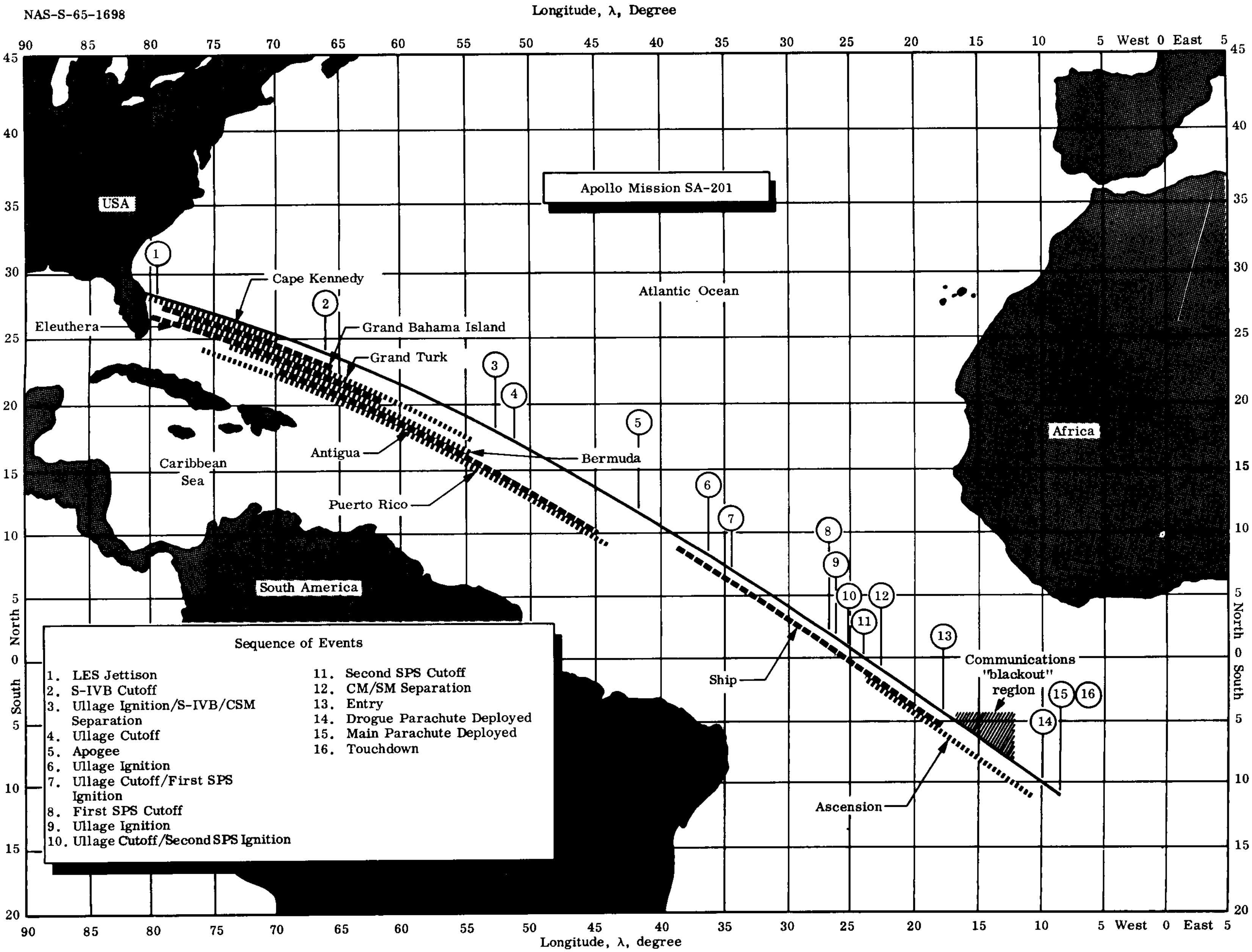
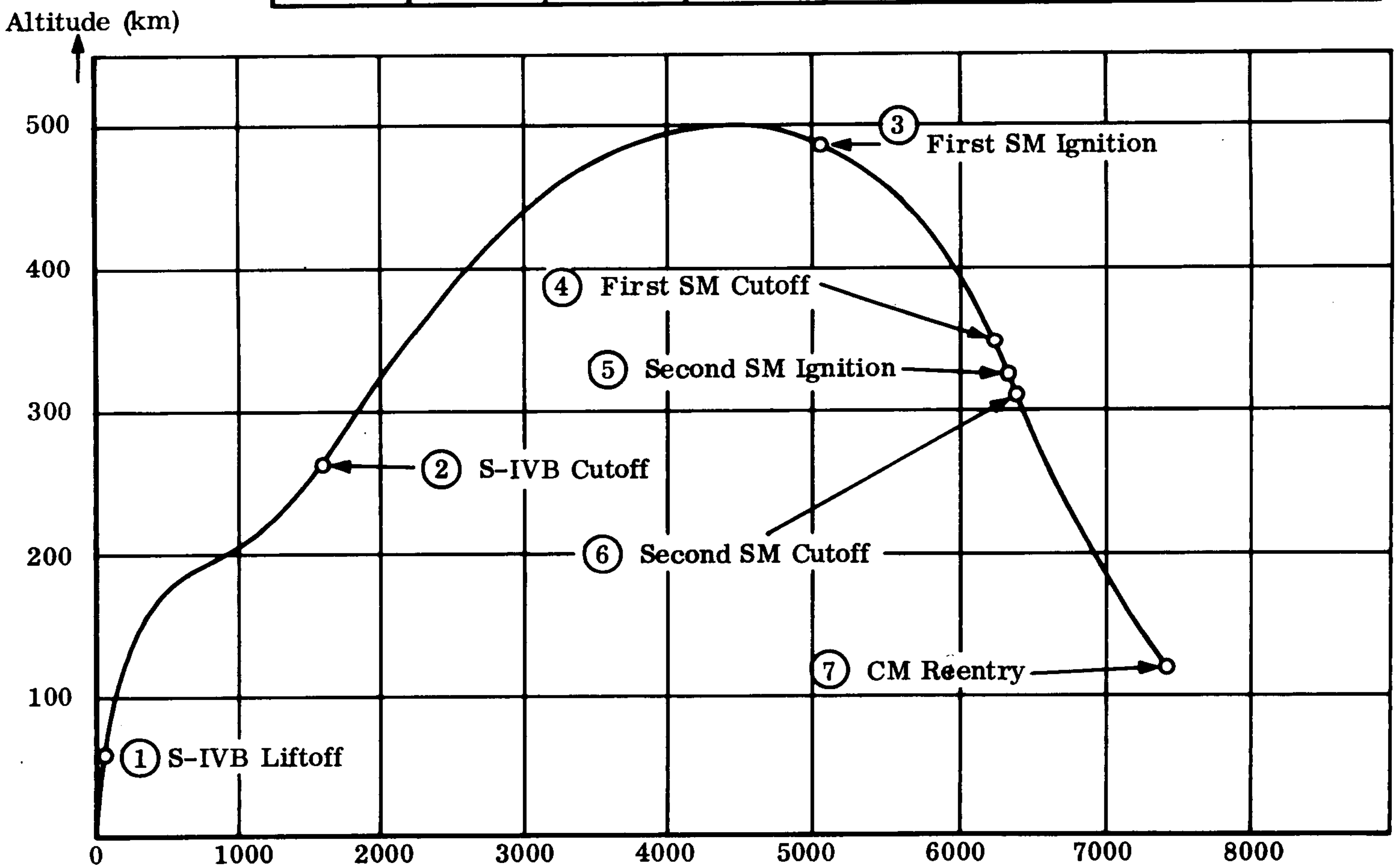


Figure 3-2. Ground Track With Noted Acquisition Tracking Range for Various Radar Stations (from "Operational Support Plan for Apollo Mission AS-201," Reference 7)

	1	2	3	4	5	6	7
1	151.75	606.85	1201.21	1381.21	1396.21	1406.21	1547.51
V	2182	6944	6647	7992	8015	8124	8388
θ	65.0	81.9	91.8	99.8	99.7	100.0	98.9



Legend: 1 – Time in Seconds
V – Velocity in Meters Per Second
θ – Inclination with Local Horizon in Degrees

Figure 3-3. Apollo-Saturn 201 Trajectory Profile (from "Saturn IB Mission Plan and Technical Information Checklist." Reference 12)

Table 3-1

Apollo-Saturn 201 Mission Profile
(References 8 and 14)

Events (A Subphase Extends From an Event to the Next Event).	Normalized Profile		
	Elapsed Time In Seconds	Phase Number	Phase Time In Seconds
Start Countdown			
		1	---
Liftoff, Hold Down Release	0.0		
		2	146.3
S-IB Cutoff	146.3		
		3	0.8
S-IB S-IVB/CSM Separation	147.1		
		4	4.8
S-IVB Engine Ignition (90% Thrust)	151.9		
		5	454.9
S-IVB Engine Cutoff	606.8		
		6	249.0
Coast & Orientation Maneuver	855.8		
		7	20.0
S-IVB/IU/SLA CSM Separation	875.8		
		8	390.2
SPS First Ignition	1266.0		
		9	180.0
SPS First Cutoff	1446.0		
		10	15.0
SPS Second Ignition	1461.0		
		11	10.0
SPS Second Cutoff	1471.0		
		12	31.5
SM CM Separation	1502.5		
		13	112.5
Entry, 0.05 G's	1615.0		
		14	425.0
Forward Heat Shield Jettison	2040.0		
		15	441.0
Touchdown	2481.0		
		16	(48 hours maximum)
Retrieval	(48.68 hours maximum)		

3.3.4 COMMAND MODULE RECOVERY PROVISIONS

For the unmanned Apollo-Saturn 201 mission, there is no need for provisions for aborts for the usual reason of crew safety. However, plans are provided for the recovery of the command module in the event of malfunction of the launch vehicle.

The profiles for the two modeled "worst case" contingencies, using the launch escape system or service propulsion system as power for separation of the command module from the launch vehicle, are shown in Table 3-2. These contingency profiles assume that a command to begin the contingency sequence is received by the flight vehicle at either 170 or 875 seconds after hold-down release.

Table 3-2

Apollo-Saturn 201 Profiles for Contingencies

Using Launch Escape System*
(Reference 8)

Event	Elapsed Time from Liftoff (Hours:	Minutes:	Seconds)
Initiation (worst case)	00	02	50
Deploy Canards	00	03	01
Jettison LES	00	03	04
Jettison Apex Cover	00	03	05
Earth Impact	00	07	34
Recovery (Maximum)	48	00	00

*A contingency using the launch escape system as power for separation can be initiated by command from the ground any time between liftoff and 22 seconds after separation of the first stage (Reference 8), at flight times between 0 and 00:02:50.

Using Service Propulsion System**
(References 8 and 11)

Event	Elapsed Time from Liftoff (Hours:	Minutes:	Seconds)
Initiation (worst case)	00	14	35
SPS Ignition	00	14	38
SPS Cutoff	00	14	48
SM CM Separation	00	15	30
Earth Impact	00	31	42
Recovery (Maximum)	48	00	00

**A contingency using the service propulsion system as power for separation can be initiated by command from the ground any time from launch escape system jettisoning until the spacecraft separates from the S-IVB (Reference 8) at flight times between approximately 00:03:00 and 00:14:35.

SECTION 4

RELIABILITY PREDICTION

4.1 GENERAL

Reliability numbers are used in specifications, predictions, and assessments. Unfortunately, they do not convey the same meaning to all people. This section explains some of the concepts used in the modeling procedure of this report.

For the unmanned Apollo-Saturn 201 mission, mission success is defined as the accomplishing of all of the mission objectives established by OMSF. These objectives are listed in paragraph 3.2 of this report.

The models used show the relationships of the systems and subsystems from the standpoint of reliability. They do not represent a functional block diagram of the hardware. Subsystems modeled in series may have no physical connection; such a series representation indicates that all must function to achieve success of the applicable activity.

4.2 MODEL SOURCES

The models used for this analysis are compiled from the reliability engineering analysis prepared by stage contractors, supplemented where necessary by similar analyses made by Apollo Reliability and Quality Assurance representatives. The reliability or unreliability of each contributing equipment or subsystem has been estimated based on past experience with similar equipments, including the effects of normal manufacture, test, and handling, and a knowledge of the design of the equipment assigned to the AS-201 mission.

The predictions resulting from this modeling activity accordingly represent the probability based on the best available data for successful operation of the AS-201 equipment as designed. They serve as baselines or references from which to evaluate the effects of variations encountered in the quality control or test of the particular pieces of hardware to be used for the AS-201 mission.

4.3 PREDICTION CRITERIA

Since the predictions (or baseline) describe only the equipment as designed, the effects of the following criteria are not included in the stated probabilities of success:

a. Qualification testing status.

b. Unusual problems in manufacturing or quality control.

c. Unusual problems in handling or test.

d. Unresolved Unsatisfactory Condition Reports (UCRs), Failure and Rejection Reports (FARRs), etc.

e. Unresolved anomalies encountered during checkout and countdown.

These effects are, however, normally considered in interpretation of the results of the mission assessments.

SECTION 5

FAILURE HISTORY

5.1 GENERAL

Failures encountered during checkout and test of the AS-200-series equipment could be used as a reliability indicator for the AS-201 mission. However, the reporting system for the AS-200-series of missions is not well established. The lag in the reporting of resolutions of unsatisfactory conditions prevents identification of reliability trends during the period when corrective action could be taken.

Since the AS-201 mission provides the first use of the failure reporting and resolution systems, it is expected that improvements in the timeliness and details of reporting will be worked out, and will be a useful tool for AS-202 and following missions.

5.2 LAUNCH COMPLEX 34 FAILURE HISTORY

Figure 5-1 shows the cumulative unsatisfactory condition reports on the Launch Complex 34 facility checkout. The rapid increase in reports during August and September coincides with increased checkout activity.

No information was available on the status of resolutions of the unsatisfactory conditions.

5.3 LAUNCH VEHICLE FAILURE HISTORY

The failure reporting systems in support of the AS-201 launch vehicle have been in a state of development up to the present time. This has made it difficult to obtain current and consistent information on the quantity, type, and status of failures encountered to date. Figures 5-2 and 5-3 depict the status of unsatisfactory conditions and their resolution on the AS-201 launch vehicle. These figures present unsatisfactory conditions encountered, beginning with post-manufacturing checkout, not just failures alone.

The first of these illustrations shows unsatisfactory conditions on a cumulative basis. The number of these failures which were resolved is shown at the end of June, August, and October. More recent data was not available.

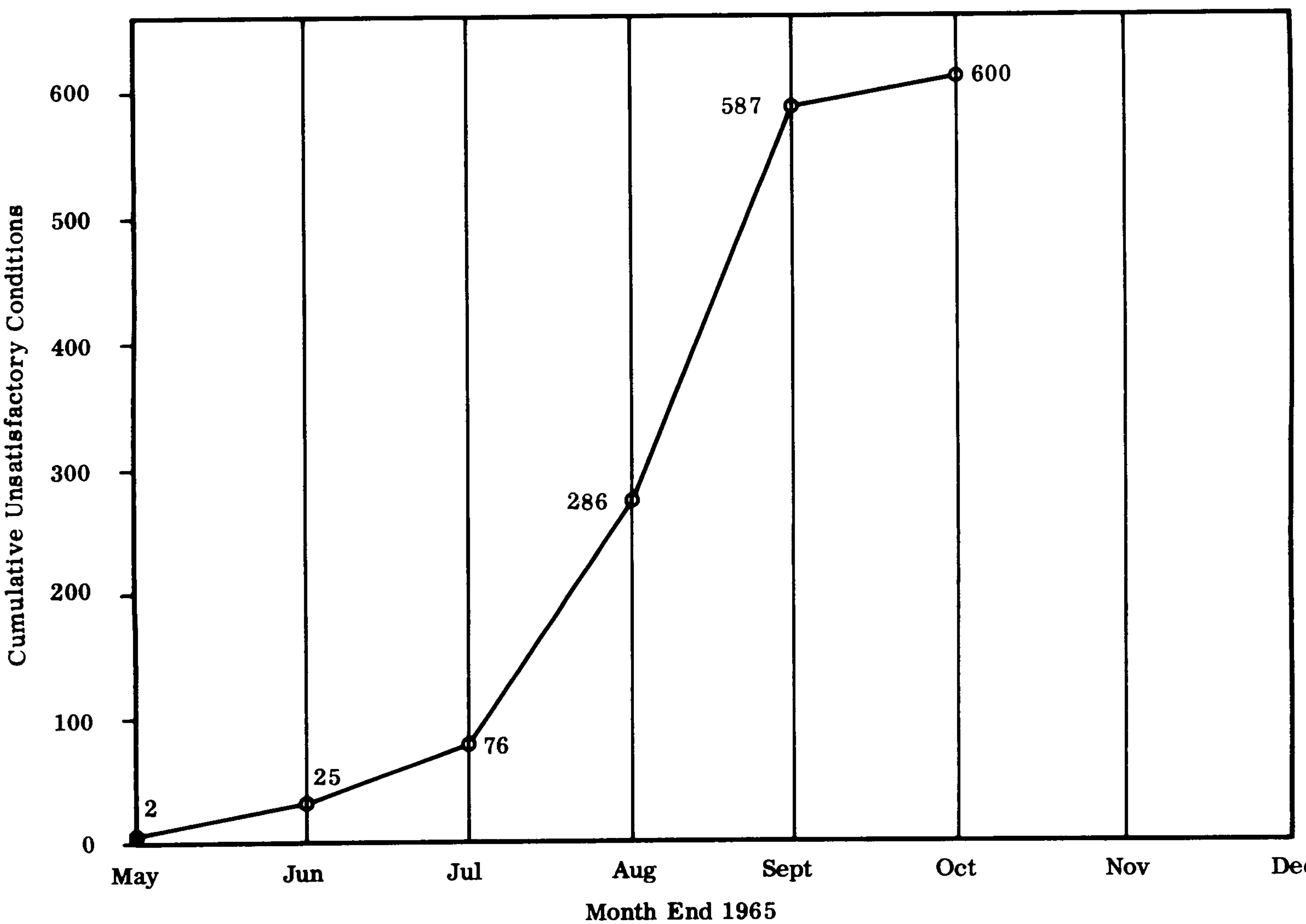


Figure 5-1. Apollo Unsatisfactory Condition Summary and Trend for LC 34 Facility Checkout

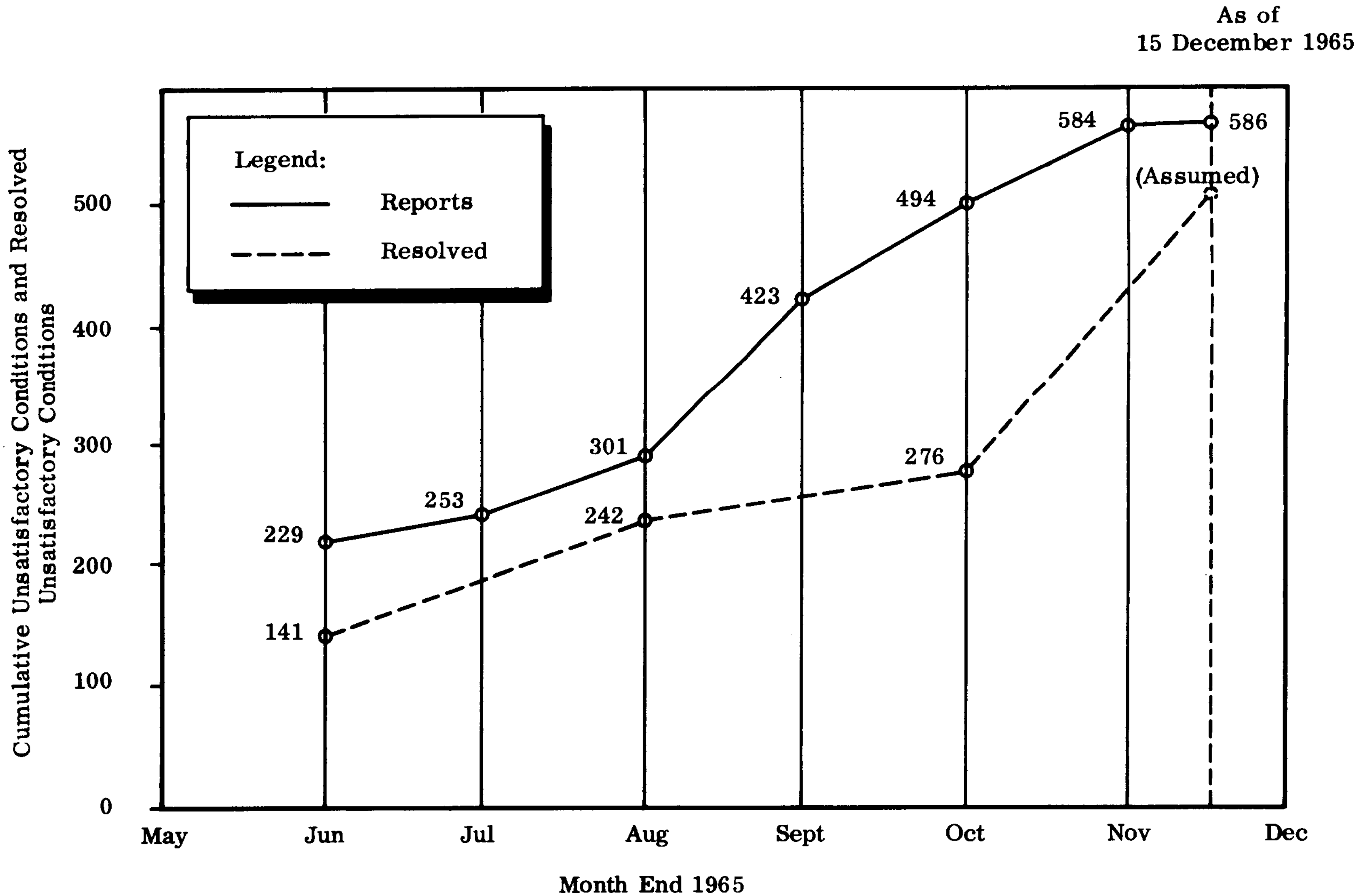


Figure 5-2. Apollo Unsatisfactory Condition Summary and Trend for Launch Vehicle AS-201

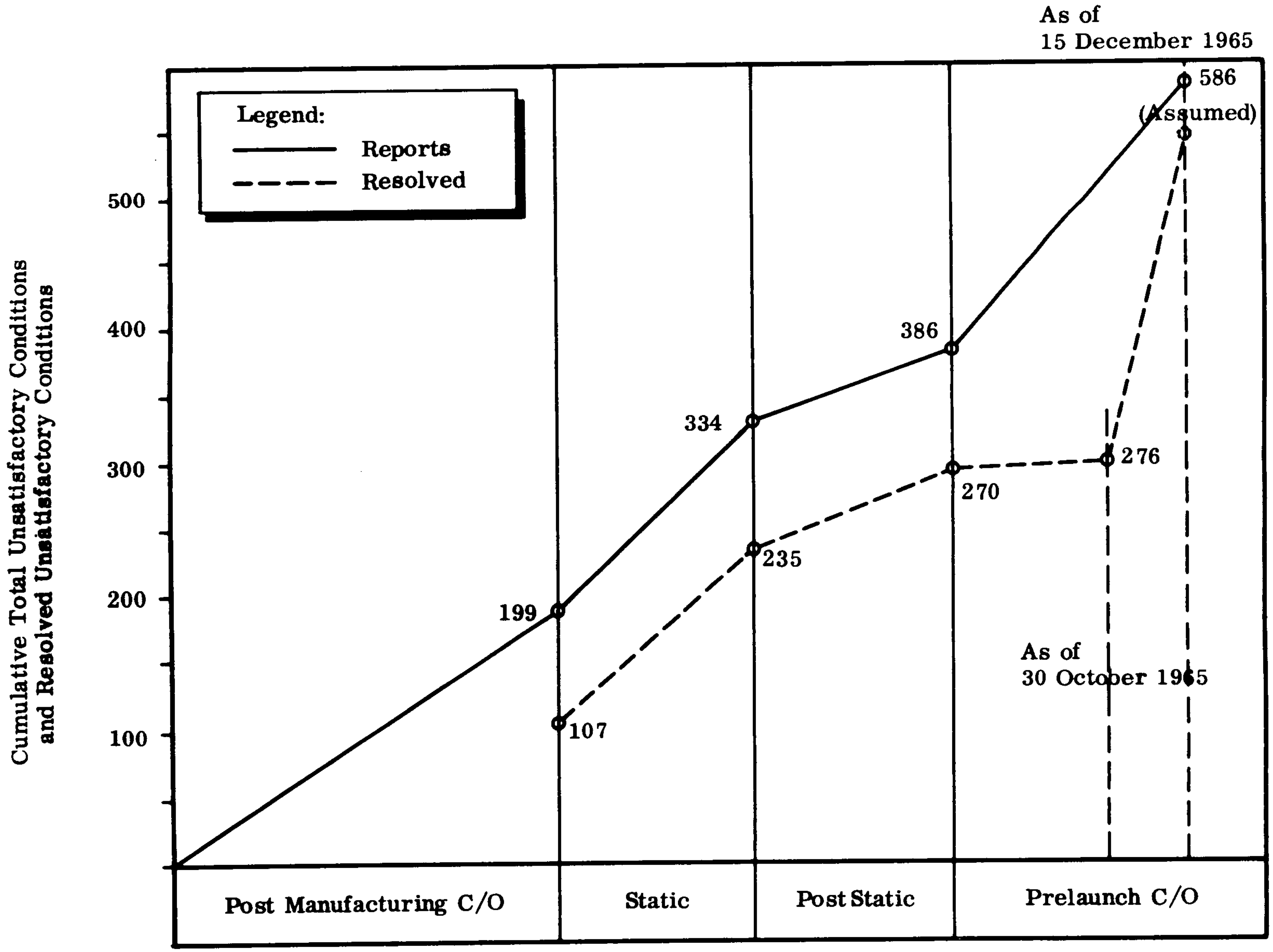


Figure 5-3. Apollo Unsatisfactory Condition Summary and Trend for Launch Vehicle AS-201

The second of these illustrations shows the buildup of unsatisfactory conditions as the launch vehicle progressed through various phases on its way to launch. The dotted line shows the ones resolved as of the end of October. In both cases the gap representing unresolved failures has been widening as the mission gets closer to launch. This is just opposite to what should happen. This condition appears to be the result of a combination of an increased incidence of unsatisfactory conditions, along with decreased incidence of resolutions. This increased incidence of unsatisfactory conditions implies that inadequate control was exercised in checkout and test of the launch vehicle prior to delivery to KSC, causing problems in launch preparations at KSC.

The recent emphasis on problems has been to "rationalize" them for the AS-201 vehicle in order to meet launch schedule rather than push corrective action to remove the problem from follow-on hardware. This shift in emphasis has also affected the operation of the data systems, and formal documentation of problems and their resolutions are falling behind. On this basis, the launch vehicle failure reporting system does not provide a positive reliability performance indicator.

5.4 SPACECRAFT 009 FAILURE HISTORY

During the early delivery phases on spacecraft 009 a computerized failure reporting system was instituted by North American Aviation and MSC. Weekly tape submittals were delivered to MSC for engineering review and resolution. Later an additional requirement was placed on the prime contractors to datafax significant failures to MSC within 24 hours of their occurrence. It was determined that failures at KSC would be datafaxed to MSC in a similar manner. After the first spacecraft reached KSC, the typical problems associated with the initiation of a failure reporting and control system were encountered.

The datafaxing procedure is designed to provide reliability and engineering data to MSC and NAA in the shortest possible time. The timeliness desired has not been reached due to the difficulty of defining and analyzing the failures.

For example, during the course of integration tests the potential failure can be in either the ground support equipment or the spacecraft. In many instances, the problem of deciding whether a particular problem constitutes a critical failure is extremely difficult. In one instance, considerable discussion took place over a 2.25-inch-long tear in a cotton sateen liner attached to a flap in the main chute. This was determined to be a failure, and remedial action was taken.

The time consumed in these analyses, unfortunately, causes a serious lag in failure reporting.

This condition is reflected in Figure 5-4. Although MSC has experienced difficulty in getting failures resolved in a timely manner, the failures which have occurred on spacecraft 009 have received special emphasis. Of the 26 unresolved failures existing when the spacecraft was shipped from Downey, 12 are resolved and the others are being rationalized.

The current status of open failures as of 4 January 1966 is:

14	Downey failures are being rationalized for Spacecraft 009
11	KSC failures under analysis
<u>42</u>	KSC failures awaiting analysis
67	Open failures at time of report

The 53 failures yet to be resolved are potential flight constraining items; however, it is expected that the analysis will be completed or the failures will be rationalized in time for the FRR.

It is concluded that failure resolution for the spacecraft is under control, and provides a positive reliability performance indicator.

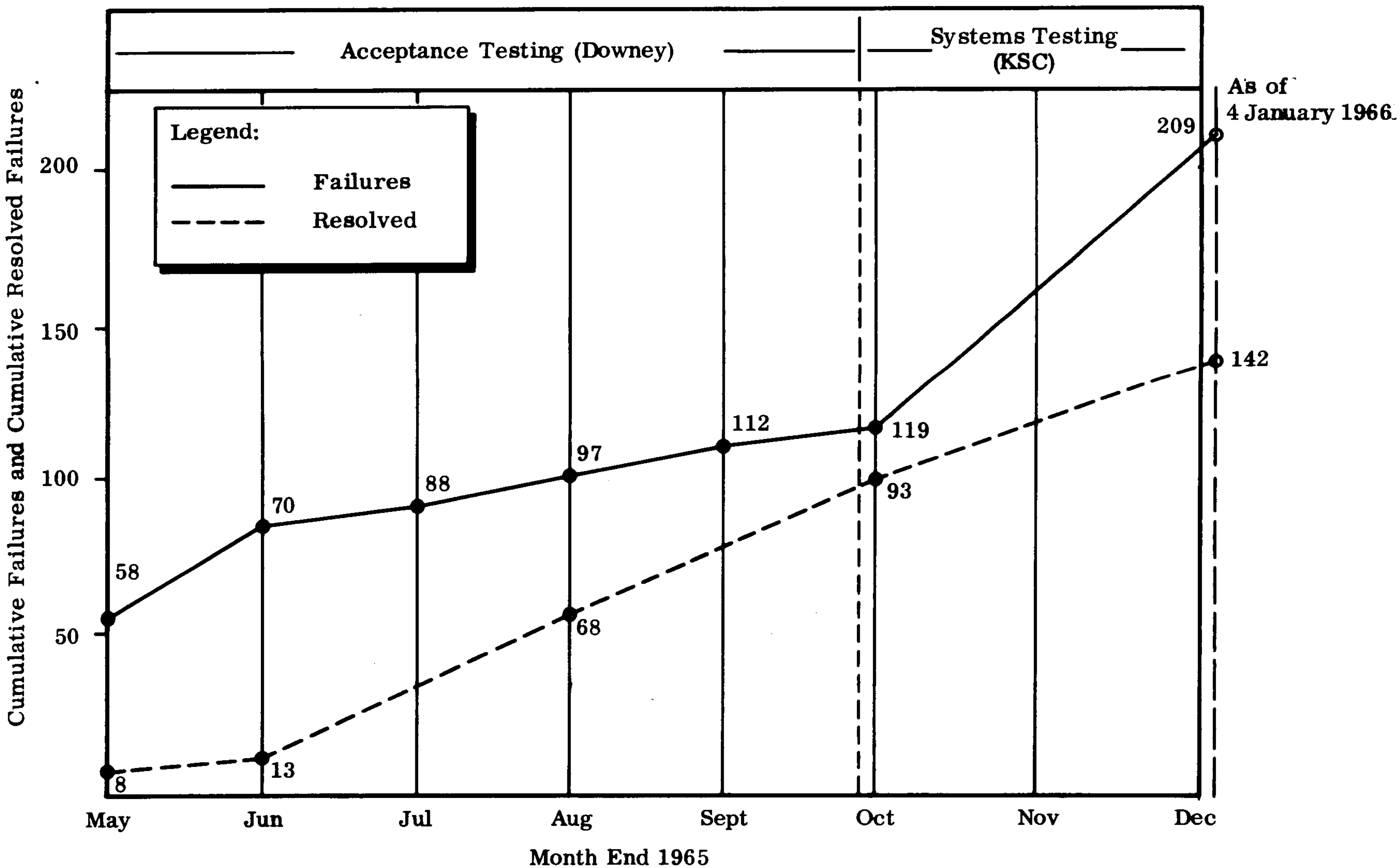


Figure 5-4. Apollo Failure Summary and Trend for Spacecraft

APPENDIXES

APPENDIX A

RELIABILITY MODELS AND TECHNIQUES

A.1 INTRODUCTION

The detailed information from which the predictions presented in this report were derived is contained in this appendix. The models used for the analysis are shown in block form, and the modeling techniques used are described.

A.2 OUTPUTS

The numerical results of the analysis are presented in this section. The information contained herein may duplicate that of the Apollo-Saturn 201 SUMMARY (Section 2), but the formats and the level of detail used are different.

A.2.1 PREDICTIONS BY MISSION OBJECTIVES

The probability of achieving each of the items in the six primary objectives for the Apollo-Saturn 201 Mission, as given in the OMSF Flight Mission Directive (Reference 1), is listed in Table A-1. These six objectives cover the demonstration, verification, or evaluation of 23 functions or subsystems. For purposes of analysis and review, the objectives were paraphrased to list these items individually.

The six-decimal-place numbers shown are indicative of the detail carried in the mathematics; two-decimal-place accuracy is considered significant for purposes of interpretation.

The unconditional or cumulative probabilities of completion include the reliabilities of all systems or subsystems needed through accomplishment of the objective; that is, the probability of the mission reaching the time when the demonstration of the objective will have been completed. The conditional probabilities are the probabilities for successful completion of the objective provided the mission has reached the time when demonstration of the objective has been started. The lack of detailed data below the subsystem level prevented the calculation of some conditional probabilities.

The techniques used for computation are described in paragraph A.5.4, and diagrams of the models used are shown with paragraph A.4.1.

Table A-1

Predictions for Objective Success

Mission Objectives (Paraphrased)	Probability of Completion	
	Unconditional	Conditional
1. Determine Structural Loading of the SLA When Subjected to S-IB Launch Environment	0.98 85 34	
2. Demonstration of S-IB/S-IVB Separation	0.98 87 48	0.99 97 70
3. Demonstration of LES Separation	0.95 93 29	0.99 94 40
4. Demonstration of LV Structural Integrity	0.95 73 83	
5. Verification of LV Propulsion Subsystem Operation	0.95 73 83	
6. Evaluate Performance of the Open-Loop EDS	0.95 74 39	0.99 99 91
7. Verification of LV Guidance and Control Subsystem Operation	0.94 97 27	0.98 80 53
8. Demonstration of LV-CSM Separation	0.94 93 70	0.99 83 81
9. Determine Long-Duration (approx 200 sec) SPS Performance, Including Shutdown	0.94 28 37	0.99 87 01
10. Demonstrate Restart of SPS Following Long-Duration Burn	0.94 28 13	0.99 86 99
11. Verification of Spacecraft SM RCS Subsystem Operation	0.94 21 49	0.99 97 74
12. Demonstration of SM-CM Separation	0.94 21 49	0.99 99 56
13. Evaluate CM Heat-Shield Performance	0.93 83 90	0.99 90 00
14. Verification of Spacecraft SCS Subsystem Operation	0.93 83 90	0.99 56 30
15. Verification of Spacecraft CM RCS Subsystem Operation	0.93 79 35	0.99 99 94
16. Verification of Spacecraft Communications	0.93 47 33	0.99 66 70
17. Verification of Spacecraft ELS	0.93 47 33	0.99 79 72
18. Verification of Spacecraft ECS	0.93 47 33	0.99 89 10

Table A-1

Predictions for Objective Success (Continued)

Mission Objectives (Paraphrased)	Probability of Completion	
	Unconditional	Conditional
19. Verification of Spacecraft EPS	0.93 47 33	0.99 95 34
20. Determine Adequacy of Recovery Aids	0.93 47 33	
21. Determine CM Adequacy for Entry From Low Earth Orbit	0.93 47 33	0.99 14 77
22. Demonstration of Spacecraft Structural Integrity	0.93 47 33	0.99 90 00
23. Demonstration of Launch Support Facilities (Launch Availability)	(First modeling of Launch Availability is planned for AS-204)	

A.2.2 PREDICTIONS BY MISSION PHASES

The predictions for mission success, according to the chosen phases of the mission profile (outlined in Table 3-1), are listed in Table A-2. The six-decimal-place numbers shown are indicative of the detail carried in the mathematics. For purposes of interpretation, only two decimal place accuracy is considered significant, and is used in other parts of this report.

Each phase of the mission begins with one event and covers the interval to the next event. The "unconditional" predictions shown are the cumulative probabilities of the mission being successful from liftoff to the beginning of each phase. The "conditional" predictions are the probabilities of each phase being completed, provided the flight has progressed to the event which starts the phase.

The technique used for computation is described in paragraph A.5.3, and diagrams of the models used are shown with paragraph A.4.2.

Table A-2

Apollo-Saturn 201 Mission Success Summary by Phases

Events (A phase extends from an event to the next event)	Predictions		
	Phase Number	To Beginning of Period (unconditional)	During Phase (conditional)
Start Countdown			
	1	(Launch Availability not modeled, assumed to be 1.0)	
Lift-Off, Hold Down Release			
	2	1.0	0.985416
S-IB Cutoff			
	3	0.985416	0.999761
S-IB S-IVB/CSM Separation			
	4	0.985180	0.999923
S-IVB Engine Ign. (90% thrust)			
	5	0.985104	0.969123
S-IVB Engine Cutoff			
	6	0.954687	0.992671
Coast & Orientation Maneuver			
	7	0.947690	0.999388
S-IVB/IU/SLA CSM Separation			
	8	0.947110	0.997994
SPS First Ignition			
	9	0.945210	0.997257
SPS First Cutoff			
	10	0.942618	0.999975
SPS Second Ignition			
	11	0.942594	0.999484
SPS Second Cutoff			
	12	0.942107	0.999962
SM CM Separation			
	13	0.942072	0.999851
Entry, 0.05 g			
	14	0.941932	0.995703
Forward Heat Shield Jettison			
	15	0.937884	0.999802
Touchdown			
	16	0.937698	0.996619
Retrieval			---
Over-all (At End of Retrieval)		0.934528	

A.2.3 PREDICTIONS FOR CONTINGENCIES (Table A-3)

A successful mission has no contingency situations. A contingency (abort) would be initiated in the Apollo-Saturn 201 mission only if a major malfunction occurs during operation of the launch vehicle stages. There are two possible contingencies for this mission, either of which will provide a nondestructive landing for the unmanned command module in an area in which recovery will be possible. Predictions have been made for the successful completion of either of these contingencies, provided it is initiated (by command from the ground) during the AS-201 flight.

Table A-3

Predictions for Apollo-Saturn 201 Contingency Success

Name of Contingency	Contingency Success	
	Odds	Probability
Launch Escape System (LES)	142 to 1	0.993
Service Propulsion System (SPS)	90 to 1	0.989

A.2.4 PREDICTIONS FOR SPACECRAFT 009

The probabilities of mission success, according to the model specially generated for Spacecraft 009, are listed in detail according to mission subphases and equipment subsystems in Figure A-1. These predictions are based on the model diagram shown in paragraph A.4.4, and the techniques and modeling details used are described in paragraph A.5.7.

The spacecraft modeling effort being conducted by MSC and its contractors is being directed initially to the AS-204 mission. The prediction model for AS-201 was generated by AR&QA to fill in a gap which would have otherwise been left in the reliability program.

A.3 INPUT DATA (See Tables A-4 through A-16)

The input data used for the model is listed in this subsection, along with the reference or source from which each value was obtained. The listing is made in the data form received (conditional probabilities of success, failure rates, etc.). The use of this data is described in other subsections within this appendix.

A.4 MODEL DIAGRAMS

The block diagrams of the models used for the analysis described in this report are included herein. These diagrams show the level of detail used in the modeling for each system and/or subsystem, and the numerical values used in obtaining the Apollo-Saturn 201 predictions.

A.4.1 PREDICTIONS BY MISSION OBJECTIVES

The diagram of the model used for the success prediction of each of the mission objectives (as paraphrased from the six OMSF objectives) is shown in Figure A-2 beginning on page A-41.

The calculations for objective success involve taking the product of the reliabilities of those subsystems actually needed to accomplish the objective (and omitting all other subsystems) through the subphase through which the mission had to proceed in order to have completed the objective requirements.

A.4.2 PREDICTIONS BY MISSION PHASES

The diagram of the model used for the success prediction of each of the chosen mission subphases is shown in Figure A-3 beginning on page A-63. For these calculations, subsystems which are required in later subphases are included in the early subphase models, since the over-all aim is for complete mission success (no alternate or partial missions were considered). For example, spacecraft subsystems appear in the "Lift-off Through S-IB Cutoff, Subphase 2" model, because they are subjected to the effects of environment and elapsed time during this subphase, and their operation is required later, even though they are passive during this subphase.

The blocks shown by solid lines represent those subsystems that are required for completion of a subphase, or are passive and will be used later. The blocks delineated by broken lines represent those subsystems that have already contributed to mission success in preceding subphases and are no longer required.

Data for the S-IB stage subsystems, as derived from the contractor's criticality analysis (Reference 18), is shown, with the comparable data from 10,000 simulated flights (Reference 19) shown in parentheses. Information at subsystem level was available (or could be generated by analogy) for the other stages. Greater detail of the Spacecraft 009 model is shown in paragraphs A.4.4 and A.5.7 of this appendix.

CONDITIONAL PROBABILITIES OF MISSION SUCCESS BY SPACECRAFT 009 SUBSYSTEMS

AS-201 Mission	EPS	SCS	ELS	LES	MDS (EDS)	SECS	SM RCS	SPS	ECS
2	0.9999945	0.9995100				1.0	0.9999890	0.9998590	0.9999500
3	1.0	1.0				1.0	1.0	1.0	1.0
4	1.0	1.0				1.0	1.0	0.9999990	1.0
5	0.9999842	0.9984600		0.9994420		1.0	0.9999670	0.9996150	0.99982
6	0.9999991	0.9999200			0.9999910	1.0	0.9999980	0.9999760	0.99999
7	0.9999999	0.9999900				1.0	0.9999800	0.9999980	1.0
8	0.9999986	0.9998700				1.0	0.9999330	0.9999630	0.99999
9	0.9999937	0.9993900				1.0	0.9999910	0.9992910	0.9993
10	0.9999999	0.9999900				1.0	0.9999880	0.9999980	1.0
11	0.9999996	0.9999700				1.0	0.9999870	0.9995480	1.0
12	0.9999999	0.9999900				1.0	0.9999770	0.9999970	1.0
13	0.9999988	0.9999600				1.0	0.9999640		1.0
14	0.9999548	0.9985800				1.0			0.99987
15	0.9999953		0.9999720			1.0			0.99999
16	0.9996160		0.9980000			0.999997			
Subsystem Reliability	0.9995343	0.9956370	0.9979720	0.9994420	0.9999910	0.999997	0.9997740	0.9982449	0.9989101

AS-201 Mission	COMM	INSTR	CM RCS	Separation	Control Programmer	Heat Shield	Impact Flotation	Structural Integrity	Subphase Total Conditional Probability	Subphase Total Unconditional Probability
2	0.9996400	0.999920	1.0		0.9999889			0.9998911	0.9987430	0.9987430
3	1.0	1.0	1.0					1.0	1.0	0.9987430
4	1.0	1.0	1.0					0.9999997	0.9999986	0.9987416
5	0.9988500	0.9997300	1.0		0.9999655			0.9996613	0.9955028	0.9942500
6	0.9999400	0.99999	1.0		0.9999982			0.9999841	0.9997953	0.9940464
7	1.0	1.0	1.0		0.9999999			0.9999986	0.9999663	0.9940129
8	0.9999100	0.99998	1.0	0.998381	0.9999971			0.9999710	0.9979942	0.9920191
9	0.9995500	0.99989	1.0		0.9999864			0.9998660	0.9972609	0.9893018
10	1.0	1.0	1.0		0.9999999			0.9999989	0.9999746	0.9892766
11	0.9999900	1.0	1.0		0.9999993			0.9999926	0.9994864	0.9887685
12	1.0	1.0	1.0		0.9999998			0.9999977	0.9999613	0.9887302
13	0.9999800	1.0	1.0	0.999956	0.9999992			0.9999917	0.9998496	0.9885814
14	0.9989100	0.99974	0.999997		0.9999678	0.9990000		0.9996835	0.9957095	0.9843398
15	0.9999000	0.99998	0.999997		0.9999967			0.9999672	0.9997981	0.9841410
16							0.999000		0.9966190	0.9808110
Subsystem Reliability	0.9966738	0.9992301	0.999994	0.998337	0.9998987	0.9990000	0.9990000	0.9990000	0.9808110	

Figure A-1. Mission Success Probabilities for Spacecraft 009 in the Apollo-Saturn 201 Mission

[redacted]

Table A-4

List of Subsystem Conditional Reliabilities

	System or Subsystem	Conditional Probability of Success	Reference*
1.	S-IB Propulsion	0.990860	18
2.	S-IB Electrical System	0.999751	
3.	S-IB Environmental Control	0.999998	
4.	S-IVB Structures	0.999890	21
5.	S-IVB Propulsion	0.985000	21
6.	S-IVB Electrical Power	0.990100	21
7.	S-IVB Thermal Conditioning	0.999998	21
8.	S-IVB Separation from S-IB	0.999770	21
9.	S-IVB Flight Control (Hydraulic)	0.997200	21
10.	S-IVB Flight Control (Auxiliary Propulsion)	0.994100	21
11.	S-IVB Data Acquisition	0.999780	22
12.	Instrument Unit Electrical Power	0.998755	27
13.	Instrument Unit Guidance and Control	0.996708	27
14.	Spacecraft Structural Integrity	0.999000	33
15.	Spacecraft Heat Shield	0.999000	33
16.	Spacecraft Impact and Flotation	0.999000	34
17.	Spacecraft Control Programmer	0.999899	33
18.	Spacecraft Electrical Power System	0.999534	26,30,33
19.	Spacecraft Stabilization and Control System	0.995637	26,31
20.	Spacecraft Earth Landing System	0.997972	29
21.	Spacecraft SM Reaction Control System	0.999774	17,25,26,34
22.	Spacecraft CM Reaction Control System	0.999994	17,25,26,34
23.	Spacecraft Service Propulsion System	0.998245	17,25,26,34

[redacted]

Table A-4

List of Subsystem Conditional Reliabilities (Continued)

	System or Subsystem	Conditional Probability of Success	Reference*
24.	Spacecraft Environmental Control System	0.998910	26,28
25.	Spacecraft Communications	0.996674	20,26
26.	Spacecraft Instrumentation	0.999230	20,26
27.	Spacecraft Sequential Events Control System	0.999997	35
28.	Spacecraft SLA-CSM Separation System	0.998381	34
29.	Spacecraft SM-CM Separation System	0.999956	34
30.	Spacecraft Launch Escape System	0.999442	34
31.	Spacecraft Malfunction Detection System	0.999991	32

*Each of the spacecraft numbers came from the especially developed "Spacecraft 009 Model." The references shown here are those used as inputs for that model. See the listings spacecraft inputs for further details.

Table A-5

Command/Service Module Electrical Power System

Equipment	Failure Rate/10^6 Hours	Reference
Service Module Battery	8	33
Motor Switch	3	33
Service Module Bus	0.01	33
Circuit Breaker	0.12	33
Circuit Interrupter	1.6	33
Main DC Bus	17	30
Inverter	140	33
AC Bus	9.4	30
Power Programmer	10	e*
Entry and P/L Battery	8	33
Bus Tie Relay	3	33
Battery Bus	1.5	e*
Battery Relay Bus	1.5	e*
Post Landing Bus	1.5	e*
Pyro Batteries	8	33
S/M Seq. Battery	8	33
Diode	0.06	33
Flight Bus	1.5	e*

*e = AR&QA engineering estimate

Table A-6

Stabilization and Control Equipment

Equipment	Failure Rate/10^{-6} Hours	Reference
Rate Gyro Package	150	31
Attitude Gyro/Accelerometer Package	260	31
Electronic Control Assembly (Pitch)	102	31
Electronic Control Assembly (Yaw)	102	31
Electronic Control Assembly (Roll)	83	31
Electronic Control Assembly (Aux)	260	31
Electronic Control Assembly (Displays)	190	31
Attitude Set/Gimbal Position Display	44	31
Delta Velocity Display	18	31

Table A-7

Earth Landing System

Equipment	Success Probability	Reference
ELS Activate	0.999999	29
25 K Baroswitch	0.999999	29
Drogue Time Relay	1.0	29
Main Time Delay	1.0	29
10 K Baroswitch	1.0	29
Apex Cover	0.999999	29
Drogue Deploy	0.999996	29
Drogue Attach	1.0	29
Drogue Disconnect	0.999992	29
Main Deploy	0.999987	29
Main Attach	1.0	29
Main Disconnect	0.998000	29

Table A-8

Launch Escape System

Equipment	Success Probability	Reference
Launch Escape Motor	0.997759	34
Igniter Cartridge	0.999000	34
Pyrogen Igniter	0.999500	34
Propellant	0.999200	34
Liner	0.999960	34
Case	0.999700	34
Nozzles	0.999400	34
Tower Jettison Motor		
Igniter Cartridge	0.999000	34
Pillet Basket	0.999900	34
Pyrogen Igniter	0.999997	34
Propellant	0.999990	34
Liner	0.999999	34
Case	0.999999	34
Nozzle	0.999991	34
Tower Separation System	0.999600	34
Explosive Bolt Assembly	0.999900	34
CM/Tower Umbilical	0.999842	34
Tower Structure	1.0	e*

*e = AR&QA engineering estimate

[redacted]

Table A-9

Malfunction Detection System

Equipment	Success Probability	Reference
Malfunction Detection System (probability of no false signal)	0.999991	32

Table A-10

Sequential Events Control System

Equipment	Success Probability	Reference
Mission Events Sequence Controller	0.99999999	35
Service Module Jettison Controller	0.99999999	35
Fluid Control Sequencer	0.999999	35
Post Landing Sequence Controller	0.999999	35

Table A-11

Service Module Reaction Control System

Equipment	Failure Rate/10^{-6} Hours**	Reference
Helium Storage	0.20	34
Helium Feed	10	34
Fuel Feed and Storage	8	34
Oxidizer Feed and Storage	8	34
Engine	4300	34

**The failure rates listed are derived from piece part estimates. For the few piece part failure rates which were not listed by NAA/S&ID (Reference 34) state-of-the-art values listed by GE/ASD (Reference 25) or AVCO (Reference 17) were used.

[redacted]

Table A-12

Command Module Reaction Control System

Equipment	Failure Rate/10^{-6} Hours**	Reference
Helium Storage	40	34
Helium Feed	13	34
Fuel Storage	58	34
Fuel Feed	0.28	34
Oxidizer Storage	59	34
Oxidizer Feed	0.28	34
Interconnect Valving	2	26
Engine	5000	28

Table A-13

Service Module Propulsion System

Equipment	Failure Rate/10^{-6} Hours**	Reference
Fuel and Oxidizer Pressurization	90	34
Fuel Feed	130	34
Oxidizer Feed	130	34
Engine Control	14	34
Thrust Vector Control	0	34
Engine	6300	34

**The failure rates listed are derived from piece part estimates. For the few piece part failure rates which were not listed by NAA/S&ID (Reference 34) state-of-the-art values listed by GE/ASD (Reference 25) or AVCO (Reference 17) were used.

Table A-14

Environmental Control System

Equipment	Failure Rate/10^{-6} Hours	Reference
Pump Assembly Check Valve	2	28
Glycol Temperature Control Valve (2.42)	2	28
Glycol Evaporator Outlet Temperature Sensor (2.23)	0.04	28
Accumulator Quantity Probe (8.16)	1.5	28
Quick Disconnect (2.26)	0.08	28
Fill and Vent Disconnect (2.24)	0.08	28
Cabin Temperature Control Valve (2.13)	2	28
Glycol Reservoir (2.29)	2	28
Pump Assembly Filter	0.01	28
Glycol Pump	40	28
Glycol Check Valve (2.33)	2	28
Glycol Temperature Sensor (2.47)	0.04	28
Glycol Evaporator (2.6)	2	28
Pump Assembly Accumulator	2	28
Glycol Temperature Control (2.22)	0.34	28
Wetness Sensor (2.44)	1.5	28
Back-Pressure Control Valve (2.39)	1.9	28
Glycol Pump Outlet Pressure Transducer (8.1)	3.9	28
Cabin Pressure Regulator (3.28)	20	28
Cabin Pressure Relief Valve (3.1)	20	28
Demand Pressure Regulator (4.16)	20	28
Re-Entry Oxygen Tank (70.1)	0.02	28
Waste Water Tank (5.15)	0.1	28

Table A-15

Communications and Instrumentation System

Equipment	Failure Rate/10^{-6} Hours	Reference
Recorder	330	20
Data Storage (Recorder)	330	20
VHF/FM Transmitter	10	20
Premodulator Processor	10	20
High-Frequency Transceiver	8	20
Signal Conditioner	3.4	20
C Band Transponder	12	20
PCM Telemetry	190	20
Audio Center	4.3	20
VHF Multiplexer	1.2	20
PAM/FM/FM/Transmitter	10	20
VHF Recovery Beacon	1.1	20
GFE Survival Beacon	1.1	20
C-Band Antenna	0	20
Scin. Antenna	0	20
Instrumentation		
Central Timing	23	20
Sensor Group A	19	20
Sensor Group B	94	20
Sensor Group C	47	20
Sensor Group Digital	38	20

Table A-16

Separation System

Equipment	Success Probability	Reference
CSM/SLA Separation	0.998381	34
Detonator	1.0	34
SLA Ordnance	0.999600	34
Splice Joints	0.999200	34
Thrusters	1.0	34
Panel Structure	1.0	34
Panel Hinge	1.0	34
SLA Structure	1.0	34
Spring Reel	0.999600	34
Shock Attenuator	0.999990	34
Umbilical Disconnect	0.999990	34
Command/Service/Module Separation	0.999956	34
Electrical Interrupter	0.99999	34
Tension Tie Cutter	0.999999	34
Umbilical Disconnect	0.999999	34

A.4.3 PREDICTIONS FOR CONTINGENCIES

The diagrams of the models used for the "worst case" launch escape system contingency and the "worst case" service propulsion system contingency are shown in Figure A-4 beginning on page A-78. Only the systems required for contingency success appear in these models, as the normal mission would be terminated if a contingency is initiated.

A.4.4 PREDICTIONS FOR SPACECRAFT 009

The detailed model generated by AR&QA for Spacecraft 009 is shown in Figure A-5 beginning on page A-81. A description of the details and modeling techniques used is presented in paragraph A.5.7 of this appendix.

"Decision boxes" (the diamond shaped symbols) are used in Figure A-5 to make the one reliability diagram represent the complete Spacecraft 009 during all flight phases of the AS-201 mission. For modeling purposes, either of two paths may be taken through a "decision box." When the path goes through the subsystem blocks, it indicates that those subsystems are active. When the path bypasses a subsystem, it indicates that use of the subsystem is not applicable.

The number near the output lines indicates the mission phases (see Table 3-1 or Table A-2 for identification of the AS-201 phases) to which a particular path applies.

A.5 DESCRIPTION OF RELIABILITY MODELS

The term "reliability models" by definition (Reference 5) includes reliability logic diagrams, reliability data, and equipment operating profiles. The modeling technique and terminology are described in detail in that reference. Some of the terms used frequently in the Apollo-Saturn 201 analysis are defined herein for this report, and details of the modeling choices made are described.

The following terms are defined by document NPC 250-1 (Reference 6) and repeated herein for convenience:

a. Reliability Prediction. An analytical prediction of numerical reliability of a system or portion thereof similar to a reliability assessment except that the prediction is normally made in the earlier design stages where very little directly applicable test data is available.

b. Reliability Assessment. An analytical determination of numerical reliability of a system or portion thereof. Such assessments usually employ mathematical modeling, use of directly applicable results of tests on system hardware, and some use of estimated reliability figures.

c. Reliability Apportionment. The assignment (by derivation from the contractual reliability requirement) of reliability goals to systems, subsystems, and components within a space system which will result in meeting the over-all contractual reliability requirement for the space system if each of these goals is attained.

d. System. One of the principal functioning entities comprising the project hardware and related operational services within a project or flight mission. Ordinarily, a system is the first major subdivision of project work. Similarly, a subsystem is a major functioning entity within a system.

A.5.1 LEVEL OF MODELING

Stage models were considered too coarse for adequate description of the mission events and for the obtaining of the predictions for successfully meeting the mission objectives. Subsystem models were used for this analysis because they provide sufficient detail for identification of the functional events required during the flight. Only mission and contingency success are considered in this analysis, since there is no crew and hence no crew safety requirements for the Apollo-Saturn 201 mission.

The Spacecraft 009 modeling used data on subsystems, black boxes, and components for synthesizing the subsystem predictions which were in turn used in the over-all AS-201 models. The Spacecraft 009 detailed modeling is representative of the modeling which is expected from the Centers and their contractors for the AS-204 mission.

A.5.2 ALLOCATION OF SINGLE-POINT DATA

The reliability input data was available in different forms. Most of the launch vehicle prediction data was received in the form of "end-point reliabilities" covering the operation of a subsystem for the entire mission. Most of the spacecraft reliability data was received in the form of "failure rates," "stress factors," etc. for the individual components which make up the various subsystems. The spacecraft subsystem conditional reliabilities were then calculated for each subphase.

In order to make analyses of the reliability of the mission according to phases and mission objectives, it was necessary to distribute unreliabilities among the parts of the mission. Unreliabilities were determined by subtracting the conditional reliabilities from one (1.0).

A single end-point prediction for the S-IB stage was used. No end-point data for the individual subsystems was received. All failures in the S-IB would occur in the first subphase of the mission, so it was not necessary to allocate unreliability across any more subphases.

End-point numbers for each of the S-IVB subsystems were received. The following assumptions were used to distribute these unreliabilities among the parts of the mission:

a. S-IVB Structures - 10 percent of failures during S-IB burn. 90 percent of failures during S-IVB burn.

b. S-IVB Propulsion - 10 percent operating time/elapsed time (percent). 90 percent during S-IVB burn.

c. S-IVB Flight Control (Hydraulic) - operating time/elapsed time (percent).

d. S-IVB Flight Control (Auxiliary Propulsion) - 10 percent through S-IVB burn, 90 percent after S-IVB burn - operating time/elapsed time (percent).

e. S-IVB Electrical - 25 percent during S-IVB burn, 75 percent operating time/elapsed time (percent) including S-IVB burn.

f. S-IVB Thermal Conditioning - operating time/elapsed time (percent).

g. S-IVB Data Acquisition - operating time/elapsed time (percent).

h. S-IB/S-IVB Separation System - fixed.

Where a subsystem was only used for a relatively short period of time a certain percentage of failures was assigned according to physical location, stresses, and operating time. One example of this is the S-IVB propulsion system, where it was assumed that 90 percent of the failures would occur during the S-IVB burn phase and 10 percent of the failures would occur according to operating time/elapsed time percentage. Operating time as used here is any time a subsystem can fail whether it is in the act of performing its function or not. If the subsystem was a fixed, one-shot item, it could only fail once and then in a given subphase. An example of this is the S-IB/S-IVB separation system which could only fail during the S-IB/S-IVB separation subphase.

For the various subsystems of the instrument unit, reliability data was received in the form of number of failures per million missions. These reliabilities were given by subphases which coincided with subphases used in this over-all analysis.

Conditional reliabilities for the spacecraft subsystems were calculated by subphase using reliability techniques, the details of which can be found in paragraphs A.4.4 and A.5.7.

A.5.3 MISSION PHASE TECHNIQUES

The subsystem subphase unreliabilities were determined from the input data. The subphase unconditional reliabilities were computed as follows:

a. From the subsystem reliability of 1.0 at liftoff, the unreliability of the system during the first subphase is subtracted. The difference is the subsystem unconditional probability for subphase 1. From this new number subtract the subsystem unreliability for the second subphase. The new difference is the subsystem unconditional probability for subphase 2. This process is continued through all the subphases.

b. The unconditional subsystem reliability for the last subphase is the subsystem probability of mission success.

This method is analogous to starting with 1000 theoretical missions and subtracting the mission failures which occur in each subphase. Thus, the original 1000 missions minus the number of failures (unreliability) occurring in subphase 1, leaves the number of missions which enter subphase 2. This number less the failures in subphase 2, leaves the number which enters subphase 3, etc.. The number remaining after the last subphase is the number of unsuccessful missions to be expected per 1000 missions launched.

Mathematically, the identical result can be obtained for any subphase, by summing the unreliabilities for all subphases up to and including the subphase in question, and subtracting this sum from the original reliability of 1.0.

After all the subsystem unconditional reliabilities were determined, the over-all subphase unconditional reliability was calculated by taking the product of all the subsystem unconditional probabilities for that subphase, since all subsystems are considered statistically independent.

Once the over-all subphase unconditional reliabilities were determined, the over-all subphase conditional reliability was calculated by dividing its unconditional reliability by the previous subphase of unconditional reliability.

A.5.4 MISSION OBJECTIVE TECHNIQUES

A subphase through which the mission had to proceed in order to have met a given objective was selected. The product of only those subsystems actually needed to accomplish the objective was calculated using the subsystem subphase unconditional reliabilities previously calculated for a mission success. This product is the probability of success in meeting a given mission objective.

A.5.5 CONTINGENCY TECHNIQUES

If major malfunctions occur during operation of the launch vehicle stages in the Apollo-Saturn 201 flight, action may be taken to separate the command module from the other parts of the vehicle, and allow it to make a soft touchdown by means of its onboard earth landing system. Propulsion for the separation may be provided by the launch escape system during the early subphases of the flight (to the time when the LES is jettisoned for a normal flight), and then by the service propulsion system (to the time of the first scheduled SPS ignition).

The command module systems required to successfully accomplish a launch escape system abort are as follows:

a. Launch Escape System (Abort Mode).
b. Structures.
c. Sequential Events Control System.
d. Control Programmer.
e. Emergency Detection System.
f. C/M Electrical Power System.
g. C/M Reaction Control System.
h. Impact and Flotation.
i. Separation.
j. Earth Landing System.

In addition, portions of the Eastern Test Range are required (but are assumed, in lieu of reliability data) to have a reliability of one (1.0).

In order to accomplish a LES contingency, the necessity is sensed by the malfunction detection system and relayed to the ground by the communications system. Ground personnel then command contingency action, which is relayed uplink and routed to the control programmer, which sends a contingency signal to the sequential events control system which, in turn, passes the signal to the launch escape system.

The reliability logic diagram for the LES contingency, Figure A-4, represents the subsystems whose operation is required for contingency success through CM recovery.

Computation of the probability of the spacecraft completing a successful LES contingency is as follows:

a. From the mission success probability prediction, take the cumulative unreliability through subphase 4 of each of the contributing systems. These values represent the unreliability up to liftoff + 152 seconds.

b. Determine the unreliability resulting from the first 18 seconds of subphase 5 (Tables A-3 through A-15). At this point the contingency is initiated.

c. Calculate the unreliability contributed during the 284 seconds from initiation to touchdown.

d. Determine the unreliability during the 48-hour recovery period.

The summation of these unreliabilities then yields the over-all unreliability, and the prediction for contingency success is obtained by subtracting this unreliability from one (1.0). Refer to paragraph A.5.3 for further details.

The spacecraft systems required to operate for a successful service propulsion contingency are:

a. Electrical Power.

b. Sequential Events Control.

c. Control Programmer.

d. Stabilization and Control.

e. Reaction Control (CM).

f. Reaction Control (SM).

g. Separation (CM/SM).

h. Separation (CSM/SLA).

i. Launch Escape (normal).

j. Earth Landing.

k. Structures.

l. Emergency Detection.

m. Communications.

n. Impact and Flotation.

o. Service Propulsion.

p. Eastern Test Range.

The Eastern Test Range is assumed to have a reliability of one (1.0) for this estimate.

Initiation of the service propulsion contingency is identical to that described for a launch escape contingency.

The reliability logic diagram for the service propulsion system contingency is shown in Figure A-4 and represents the spacecraft systems whose operation is required through CM recovery.

Computation of the probability of completing the SPS contingency involves:

a. Determination of the system unconditional reliability at the end of subphase 7.
b. Determination of the system unreliability from contingency initiation (end of subphase 7) to recovery.
c. Sum a. and b. for total unreliability, and subtract this from one (1.0) to obtain the prediction for contingency success. Refer to paragraph A.5.3 for further details.

A.5.6 LAUNCH VEHICLE MODELS

Logic diagrams and data furnished by MSFC and its contractors were used for the Apollo-Saturn 201 launch vehicle. General S-IB information was available from the Chrysler Corporation (Reference 18). S-IVB models were provided by the Douglas Aircraft Company (Reference 21). Preliminary instrument unit data was obtained from International Business Machines (Reference 27). These models were considered adequate for inputs to the over-all Apollo-Saturn 201 model, so no areas models were generated for the launch vehicle stages.

A.5.7 PREDICTION MODELS FOR SPACECRAFT 009

A description of reliability prediction inputs for the spacecraft is contained in this paragraph. The results of the computations are described in paragraph A.2.4.

Contractor/Center predictions pertaining to this spacecraft were not made, but a functional assessment had been accomplished based on projected equipment and system tests and an assumption of no unexplained failures (Reference 69).

The prediction model generated for the Apollo-Saturn 201 analysis used the following technique:

a. The mission profile used is shown in Table 3-1 of the report.

b. Reliability logic was developed for each subsystem, with equipment configuration representing that required to successfully accomplish each subphase. In most cases, the reliability logic was contractor data published against Block I or Block II spacecraft systems. Modifications made the logic compatible with the mission profile and ground rules. When no contractor logic was available, other system information or the latest contractor end-point reliability value was used, with the unreliability being allocated among the several subphases.

c. Contractor supplied failure rate data was used where available; otherwise, data was extracted from Apollo Program Office data bank. In a few cases, equipment was assumed to have a reliability equal to 1.0.

d. Environmental modifying factors generated by Grumman Aircraft Engineering Company were applied during all boost phases and re-entry to account for reliability degradation due to high g-loading, high heat transfer, vibration, etc. The factor used was 10 for boost and re-entry periods; it was 1 at all other times.

e. Probabilities of mission success, mission objectives, and contingencies were computed as described in paragraphs A.5.3, A.5.4, and A.5.5.

f. Equipment not required for mission success during a subphase was assigned a reliability of 1.0 during that subphase.

The reliability model used for computing the probability of mission success of spacecraft 009 is shown in Figure A-5. In order to simplify the presentation of the model, decision boxes (see paragraph A.4.4) are shown which identify success paths during any mission subphase. System configuration is discussed in the following paragraphs.

A.5.7.1 Electrical Power System

The electrical power system is essentially identical to the Block I configuration. The major exception is that fuel cells have been replaced by three, 40-ampere-hour batteries. A power programmer also has been included which automates electrical power system controls normally operated by the flight crew.

The power system configuration is established prior to liftoff (that is, all loads are connected). Normal and abort enabling configuration changes which occur during flight and recovery periods are controlled by the power programmer. The energy sources consist of:

a. Three - 40 ampere-hour batteries
b. Two - 25 ampere-hour entry batteries
c. One - 25 ampere-hour post-landing battery
d. Two - 75 ampere-hour pyro batteries

The primary power source for the spacecraft loads comes from three, 40-ampere-hour batteries located in the service module, which will supply all loads except pyro until service module separation. At that time the entry batteries become the prime power source. Following earth impact the post-landing battery and the entry batteries are used.

DC is converted to AC by the use of three inverters and distributed by two redundant AC buses, plus the necessary control, protective equipment, connectors, wiring, etc.

Reliability logic shown in Figure A-5 pertaining to the electrical power system was generated from the North American schematic drawing (Reference 36). Failure rates used for each equipment are shown in Tables A-3 through A-15. A stress factor of 10 was applied during subphases 2, 5, 9, 11, and 14 to account for high g-loading, high heat transfer, and vibration encountered during re-entry.

Several ground rules influenced the construction of the electrical power system reliability model.

a. The normal mission demand is such that mission success requirements are met with two of the three 40-ampere-hour batteries operational.
b. The re-entry loads require both 25-ampere-hour batteries.
c. The post-landing loads are such that one of three batteries is sufficient during the recovery period.
d. The two pyro batteries supply redundant, electrically isolated, pyro circuits. Only one pyro battery must operate at any one time.
e. One of three inverters must operate.
f. The power programmer is required during the entire mission.

A. 5. 7. 2 Stabilization and Control System

The stabilization and control system includes the following Block I (Series J) components:

a. Rate gyro package and mounting base.
b. Attitude gyro and acceleromater package and mounting base.
c. Electronic control assembly, pitch.
d. Electronic control assembly, yaw.
e. Electronic control assembly, roll.
f. Electronic control assembly, auxiliary.
g. Electronic control assembly, displays.
h. Attitude set/gimbal position display.
i. Delta velocity display.

The backup capability to control spacecraft attitude from the ground, using telemetered data from the Wyanco attitude reference system, was deleted (Reference 9). The revised configuration is fully dependent on successful operation of the stabilization and control system; this is reflected in the logic diagrams incorporated in the present analysis by placing the components in series.

The logic diagrams for the stabilization and control system, shown in Figure A-5, incorporate the requirement for all components to function from liftoff to the end of the second service propulsion system burn. Functions of this system beyond that time include only attitude reference and control to orient the spacecraft for re-entry, and attitude stabilization and control during re-entry to the point of forward heat shield jettison.

The failure rates used for the analysis derived from North American Aviation's quarterly report (Reference 31). These failure rates shown in Tables A-3 through A-15 are from the latest procurement specifications for the Block I stabilization and control system.

A.5.7.3 Earth Landing System

The earth landing system is a Block I configuration consisting of two drogue and three main parachutes plus associated mortars, pyrotechnics, and miscellaneous hardware; two identical earth landing sequence controllers; and the apex cover. The functions performed by the earth landing system are essential for the success of either a normal mission or a contingency. They are apex cover jettison, drogue parachute deployment, drogue release, main parachute deployment, and main parachute disconnect. The functions of these components are as follow:

a. The earth landing sequence controller provides the automatic sequencing required. Either of two controllers can initiate earth landing functions, except main parachute disconnect which requires both. Only one baroswitch is required to operate.

b. The apex cover (forward heat shield) provides a protective covering for the parachute deck. It is jettisoned, prior to deployment of parachutes, by initiation of four gas-actuated thrusters.

c. The drogue parachutes orient and decelerate the command module for safe deployment of the main parachutes. The drogues are deployed in a reefed condition and at the end of eight seconds are unreefed allowing full opening.

d. The drogues are released at the time of main chute deploy. A linear shaped charge disconnects the two drogues.

e. Three parachutes are provided, any two of which are sufficient for safe landing. Pilot chutes are mortar deployed and drag out the main chutes, which unreef after eight seconds.

f. After touchdown, the post-landing sequence controller and both earth landing sequence controllers initiate two linear shaped charges which disconnect the main parachutes from the command module.

The reliability logic shown in Figure A-5 was developed on a functional logic basis from North American's detailed model (Reference 29). The numerical values used are shown in Tables A-3 through A-15. No environmental modifying factors were used since contractor fixed-point numerical values were applied in the estimate computation.

A. 5. 7. 4 Launch Escape System

The launch escape system is a Block I configuration consisting of:

a. Structures.
b. Launch escape motor.
c. Tower jettison motor.
d. Pitch control motor.
e. CM/tower separation.
f. CM/tower umbilical.
g. Canard subsystem.

The mission success requirement of launch escape system jettison requires the structure, tower jettison or launch escape motor; command module-tower separation; and umbilical functions to operate. The functions of these components are as follows:

a. The LES structure consists of the launch escape tower, the boost protective cover (which protects the forward and crew compartment heat shields), and the ballast enclosure (which covers the ballast required to establish the proper center of gravity).
b. The launch escape motor and the pitch control motor provide power for all LES contingencies.
c. The tower jettison motor carries the launch escape tower away from the CM after LES contingencies are no longer allowed; if it should fail, the launch escape motor can jettison the tower.
d. The umbilical includes all the firing, monitoring, and control circuits between the command module and the launch escape tower.

The reliability logic diagram for the launch escape system is contained in Figure A-5. The fixed-point reliability values used are shown in Tables A-3 through A-15. No environmental modifying factors were used.

A. 5. 7. 5 Malfunction Detection System

The malfunction (emergency) detection system (MDS) has sequencing and voting logic contained in the mission event sequence controller and detection and monitoring equipment contained in the instrument unit and the booster stages. The MDS monitors booster engine performance and generates contingency signals when malfunctions occur. In manned flights automatic aborts may be triggered by the MDS until automatic abort features are disabled, just prior to first stage burnout.

For mission AS-201 the MDS operates in an open-loop mode with the automatic abort enable switch in the OFF position throughout the mission. During normal booster-powered flight the MDS produces no output. Loss of thrust in two of the eight first stage booster engines or excessive angular rates of pitch, yaw, or roll will produce contingency signal to the spacecraft voting logic. During normal flight, test signals to the spacecraft will be generated when the first stage engines burn out and during second stage burn.

Performance of the MDS after first stage booster cutoff is the same for this flight as for manned flights except for the omission of onboard displays. Abnormal second-stage booster performance will be telemetered to ground control where contingency decisions may be made.

As the MDS operates open-loop, its performance is not vital to other mission objectives except under two conditions:

a. If contingency situation occurs and no signal is telemetered. Under this condition total mission failure could result.
b. If false contingency signal is generated and is not recognized as false by ground control. Under this condition an otherwise successful mission could be prematurely terminated in this case.

A fixed-point reliability value was used for the malfunction detection system. This value, listed in Tables A-3 through A-15, was furnished by North American Aviation (Reference 32). It is the probability of no false abort signal being generated, and is representative of the mission success configuration of the system.

A.5.7.6 Sequential Events Control System

The sequential events control system installed on spacecraft 009 includes the following Block I equipments:

a. Master Events Sequence Controller (MESC).

b. Fluid Control Sequencer (FCS).

c. Service Module Jettison Controller (SMJC).

d. Post-Landing Sequencer Controller (PLSC).

Redundant equipments are installed for each of the above except for the post-landing sequence controller. The earth landing sequence controller has been included with the earth landing system. The control programmer is considered separately.

The master events sequence controller receives inputs from the control programmer and provides inputs to the fluid control sequencer, the service module, the jettison controller, the post-landing sequence controller, and the earth landing sequence controller. Its performance is essential to mission success although certain functions can be bypassed by the control programmer.

The fluid control sequencer provides capability to dump the RCS propellants after a high-altitude contingency or a normal mission. Control is by manual switch on manned flights and automatic on this mission. The failure of the FCS could cause mission failure as a result of explosion and fire if RCS propellants are not discarded.

The service module jettison controller controls events in the service module to avoid collision courses after CM/SM separation. Once separation is accomplished, failure of this subsystem is considered a mission failure, since a collision of the SM and CM may result.

The post-landing sequence controller controls events after landing. These are main chute disconnect, inflation of the flotation bags, and deployment of recovery aids. The post-landing sequence controller and both earth landing sequence controllers are required to disconnect the main chutes since one of two attach points is broken by each earth landing sequence controller. The flotation bags are required only if an apex-down position is assumed by the CM after a water landing. Recovery aids may be required, depending on the location of touchdown.

Reliability logic diagrams applicable to the various subphases are shown in Figure A-5. No attempt was made to model elements of the sequential events control system below the controller or "black-box" level. Failure predictions for the individual controllers were obtained from North American Aviation (Reference 35).

Individual controllers were modeled only if their proper operation was necessary for the successful completion of a subphase. An environmental modifying factor of 10 was applied in those subphases experiencing high g-loading, heat transfer, and vibration (subphases 2, 5, 9, 11, 14).

A.5.7.7 Service Module Reaction Control System

The logic diagrams show only equipment in use in each subphase. For most of the mission the model consists of helium storage and feed and propellant storage and feed. The engine is in the logic diagrams only for actual burn time.

Contractor data was directly applied to the logic diagram elements, supplemented by state-of-the-art data. Contractor data was occasionally in the form of failures/cycle. Cycles for the purpose of this analysis were defined to be equipment actuations. The positive expulsion tank is shown with a failure rate of 1×10^{-6}/hour. Bell Aerosystems had previously shown a failure rate of 13000×10^{-6}/expulsion cycle at a rate of 30 cycles per second in the pulse mode. The contractor number was used, but the large difference requires explanation. The engine presently has the highest failure rate (4323×10^{-6}/hour). Logic diagrams were defined for the mission events (Reference 34).

A.5.7.8 Command Module Reaction Control System

Equipment not used during a subphase is not included on the logic diagrams. For the purposes of this analysis either reaction control subsystem could produce mission success; however, simultaneous A and B reaction control system operation eliminates the "switchover" requirement. Contractor logic diagrams were used in the analysis.

The failure rate data used for the CM/RCS was contractor supplied. Most of the data is Block II, supplemented by state-of-the-art data. Cyclic data (failure rate per cycle) was used to determine correction factors for the time-dependent failure rates. In general, initiation of an operation was assumed to be the start of a cycle. Contingency models are not considered for the RCS on this mission. Although the system is not pressurized until subphase 13, the storage system includes the interconnect valves in the pressurized lines; hence, their inclusion in the logic diagrams prior to subphase 13.

The large RCS engine failure rate (5000 x 10^{-6}/hour), when applied to 10 engines during a long burn phase results in a low subphase reliability. The next most unreliable phase is the re-entry phase, due in part to the high stress environment. The most unreliable items are the:

a. Engines.
b. Helium Storage Squib Valves.
c. Fuel Storage Squib Valves.
d. Oxidizer Storage Squib Valves.

A.5.7.9 Service Propulsion System

The service propulsion system is basically a Block I configuration and includes the following subsystems:

a. Fuel and Oxidizer Pressurization.
b. Fuel Feed.
c. Oxidizer Feed.
d. Thrust Vector Control.
e. Engine Control Section.
f. Engine.

All subsystems are considered to be in series for the entire mission because all are system essential. The thrust vector control subsystem consists of the entire gimbal assembly. The fuel and oxidizer pressurization, fuel feed, oxidizer feed, and engine control subsystems include the components necessary for the firing of the SPS. Thrust vector control is not initiated until T + 1200 seconds.

The logic diagrams for the service propulsion system incorporate the requirement that all systems function from liftoff to the end of second SPS burn. Thrust vector control failure data was given in cycles; therefore, the reliability estimate calculated was based on number of times used.

Failure rates used for this analysis are derived from state-of-the-art and latest available contractor estimates for the Block I configuration.

A.5.7.10 Environmental Control System

The command module for mission AS 201 is equipped with a partial Block I environmental control system. It includes the entire water-glycol loop except the space radiators and their flow controls, the cabin pressure regulator and relief valve, the demand pressure regulator, the re-entry oxygen tank, and the waste water tank.

The ECS reliability logic diagram is shown in Figure A-5. All components must perform satisfactorily throughout the mission and are, therefore, series items.

Failure rates for individual components were obtained from Reference 28.

A.5.7.11 Communications and Instrumentation

The system is composed of equipment necessary for transmitting real-time telemetry, C-band radar tracking signals and recovery beacon signals to the Eastern Test Range and recovery forces, and also equipment for receiving command data. The instrumentation subsystem includes equipment to:

a. Detect, measure, and condition information required for mission objectives.
b. Aid in real-time operation.
c. Determine environments and performance of the spacecraft subsystem.

The logic diagram used in this analysis has all the operating equipment in series for mission success. For an unmanned mission of short duration, the equipment is considered operating for the full mission. Environmental stress factors were applied to component failure rates during boost phases.

A.5.7.12 Separation System

The separation system consists of the command/service module and the spacecraft/LEM adapter separation subsystems, both of which must function.

Spacecraft/LEM adapter separation is accomplished on command from the mission event sequence controller by means of ordnance devices. Redundant detonators are detonating fuses installed around the adapter panels to effect separation. Simultaneously, two umbilical disconnects are actuated, disconnecting all wiring between the spacecraft and adapter. Thrusters on the adapter panels are actuated causing them to open. A negator spring reel then retracts each panel back over the instrument unit.

Command/service module separation is effected by cutting three tension ties, interrupting electrical circuitry, and severing the umbilical.

Spacecraft/LEM adapter separation reliability logic is shown in Figure A-5 and is derived from Reference 34 as is the command/service module separation reliability. Fixed-point reliability values are used. No environmental modifying factors were applied.

A.5.7.13 Control Programmer

The Control Programmer consists of:

a. Automatic Command Controller.

b. Sequential Timer.

   (1) Normal Mission Timer.

   (2) Abort Timer.

c. Attitude Reference.

The automatic command controller receives signals from the sequential timer to perform switching functions that on manned flights would be performed by the crew. Should the automatic command controller fail to perform a switching function, the radio command controller can accomplish the switching upon command from the ground via the radio command receiver and VHF/2-KMC antennas in the communications system. Some radio command controller outputs bypass other spacecraft sequencers and go directly to the affected system; for example, the reaction control system propellant valves.

Accordingly, the radio command controller is redundant (in certain of its channels) to both the automatic command controller and the normal spacecraft sequencers.

The attitude reference is an inertial reference unit borrowed from the Little Joe project. Its only present function, not mission-essential, is to provide an attitude history for post-flight data analysis via telemetry.

The control programmer was modeled as a single element with a fixed-point reliability (Reference 33).

The control programmer was assumed to function from liftoff to touchdown. The conditional reliability for each subphase was computed by apportioning the single-end-point number (0.99989) among the subphases in proportion to the effective operating time (product of environmental adjustment factor and subphase time interval) for each subphase.

A.5.7.14 Heat Shield

The prime objective of Mission AS-201 is evaluation of the heat shield. A fixed-point reliability value was input to the estimate computation. This value (0.999) was obtained from Reference 33.

A.5.7.15 Earth Impact and Flotation

The earth impact and flotation system provides for impact shock attenuation and for uprighting the command module. The uprighting subsystem consists of a compressor and three flotation bags which inflate in sequence to upright the command module.

A fixed-point reliability value was used for the earth impact and flotation system following touchdown. This reliability value (0.999000) was obtained from Reference 34.

A.5.7.16 Structures

A.5.7.16.1 Command Module Structure

The command module structure is made up of an inner structure or pressure vessel, and an outer structure or heat shield. The inner structure is the primary load-carrying structure of the command module during flight.

The CM heat shield is required to hold the internal temperature to 200°F or less during re-entry. It is a three-piece structure consisting of the following:

a. A forward heat shield, which covers the apex of the spacecraft.
b. A crew compartment heat shield which forms the remainder of the conical portions.
c. An aft heat shield, which covers the aft or blunt portion of the spacecraft.

A.5.7.16.2 Service Module Structure

The primary structure of the service module is a 155-inch-long cylindrical shell made up of aluminum, honeycomb-sandwich panels one inch thick. The internal structure of the service module (SM) consists of six radial beams, a forward bulkhead, and an aft bulkhead.

Three tension ties attach the CM to the SM; explosive charges separate the tension ties when the command module is to be released.

A.5.7.16.3 Spacecraft-Launch Vehicle Adapter

The spacecraft-launch vehicle adapter joins the service module to the instrument unit and, in future missions, will enclose the lunar excursion module (LEM).

The upper portion of the adapter (approximately 21 feet) is composed of four panels. Mild-detonating-fuze (MDF) explosive trains are located around the periphery of each panel. When the service module is separated from the adapter, these panels are separated by the shaped charge. Each panel is strap-hinged to the lower portion of the adapter and, at separation, the panels are rotated about the hinge line by four gas generator thrusters. A cable attached to each segment prevents the panels from moving beyond 45 degrees. The lower portion of the adapter remains in one piece attached to the instrument unit.

For the purposes of this estimate, a fixed-point assessment of 0.999 was used (Reference 33). The reported assessment was based on a marginal analysis and is essentially a measure of the safety factor designed into the structure against predicted loads.

A.5.8 CONVERSION OF CRITICALITY NUMBERS

S-IB data was supplied by the contractor as criticality numbers (Reference 18). The procedure specified by Reference 15 was used for converting these criticality numbers to success probabilities. These probabilities were then used in the AS-201 analysis. The details of the conversion are described in the following paragraphs.

A summation of all the criticality numbers for the critical items comprising each subsystem was made. The critical items list tabulated all the components that would actually or possibly cause a loss of mission or loss of vehicle after liftoff, with their respective criticality numbers. The quality of components within a given subsystem was taken from the failure effects analysis sheets. Where more than one of a given item was used, the criticality number was multiplied by that quantity.

After the sum of all the criticalities was found, the subsystem's reliability (R) was determined by:

$$R = 1 - \Sigma \text{criticality numbers} \times 10^{-6}$$

This equation is adequate for closely approximating a predicted reliability for highly reliable systems.

The system reliabilities were then calculated by taking the product of the subsystem reliabilities, and the over-all stage reliability was calculated by taking the product of the system reliabilities.

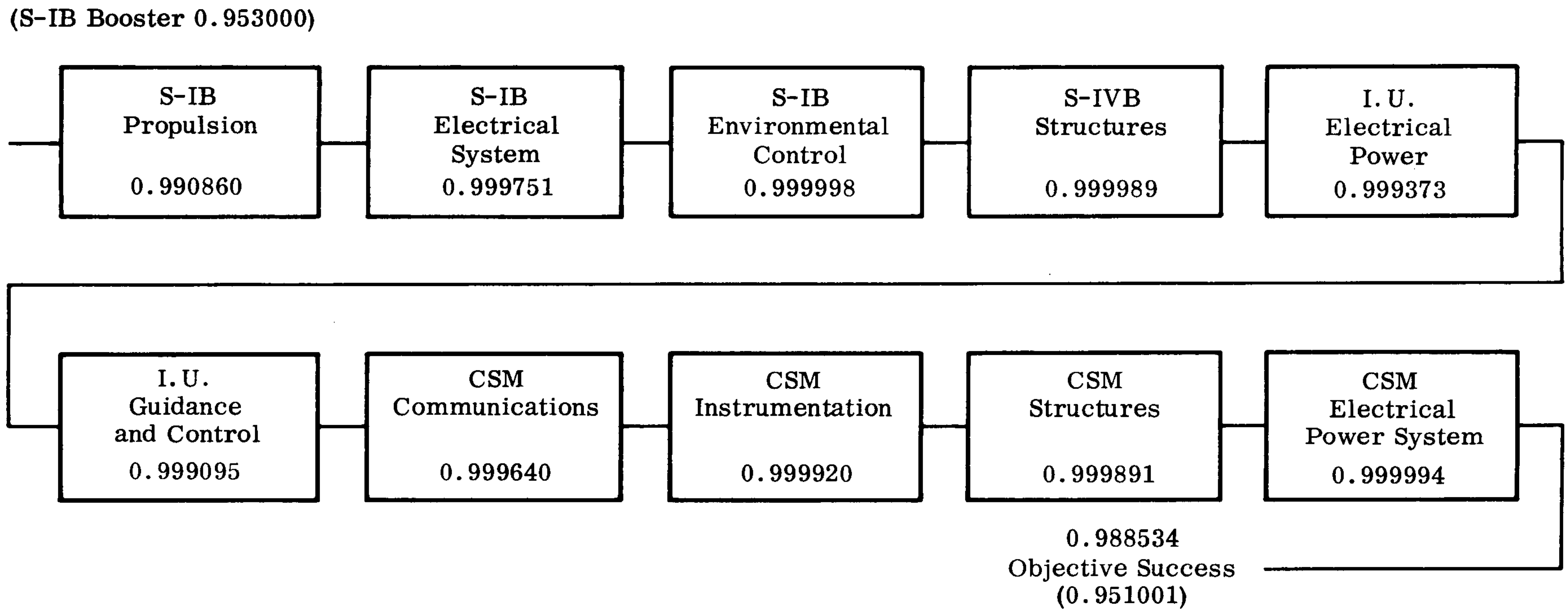


Figure A-2(a). Determination of Structural Loading of the SLA When Subjected to S-IB Launch Environment

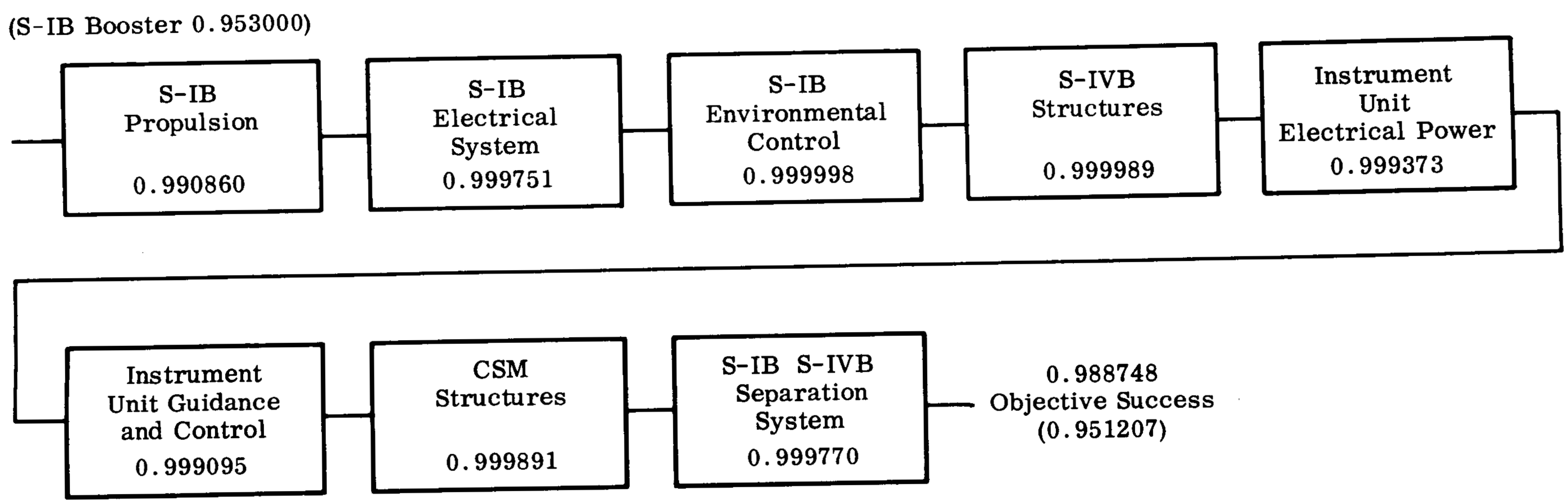


Figure A-2(b). Determination of S-IB S-IVB Separation

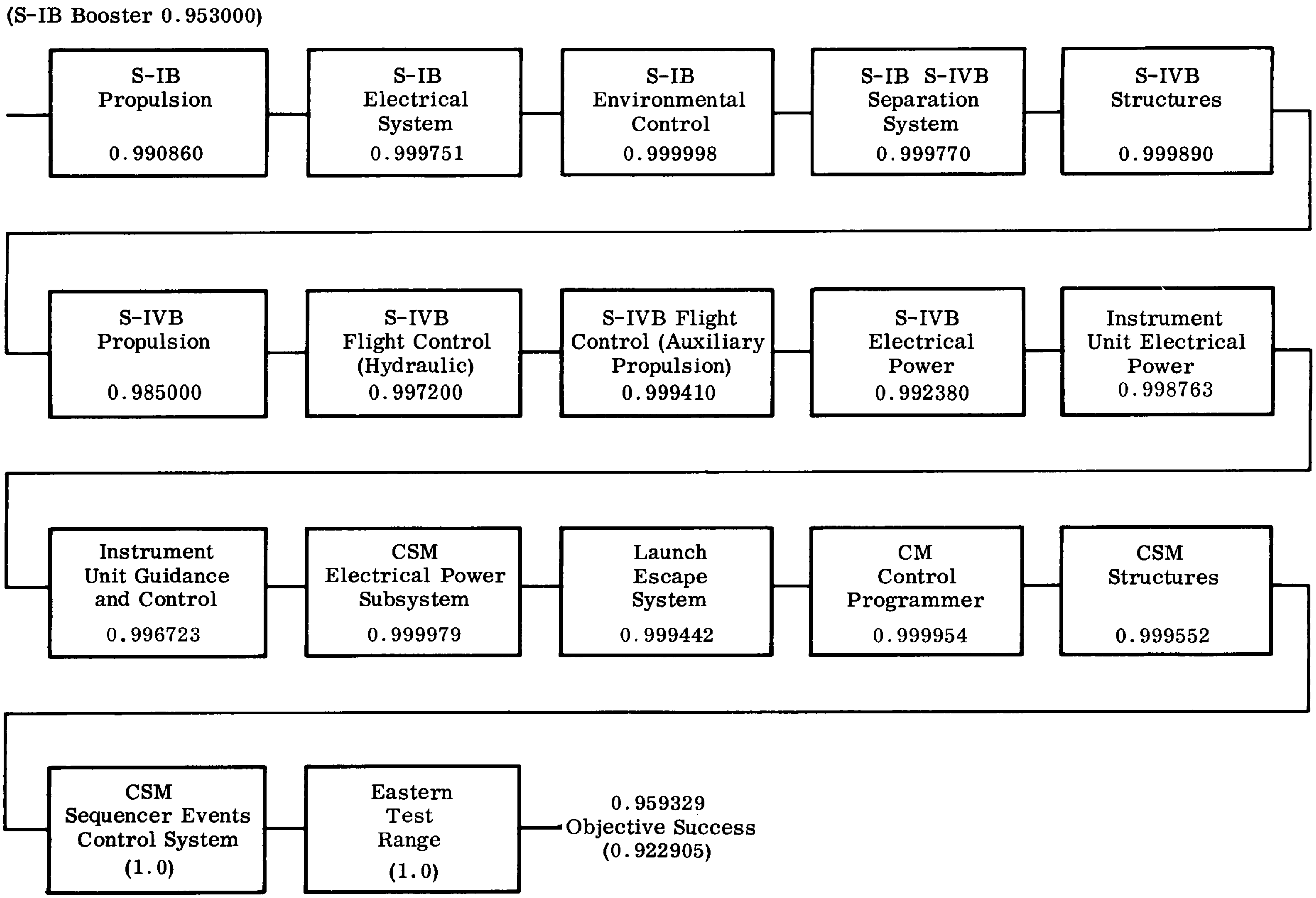


Figure A-2(c). Demonstration of Nominal Mode Separation of Launch Escape System

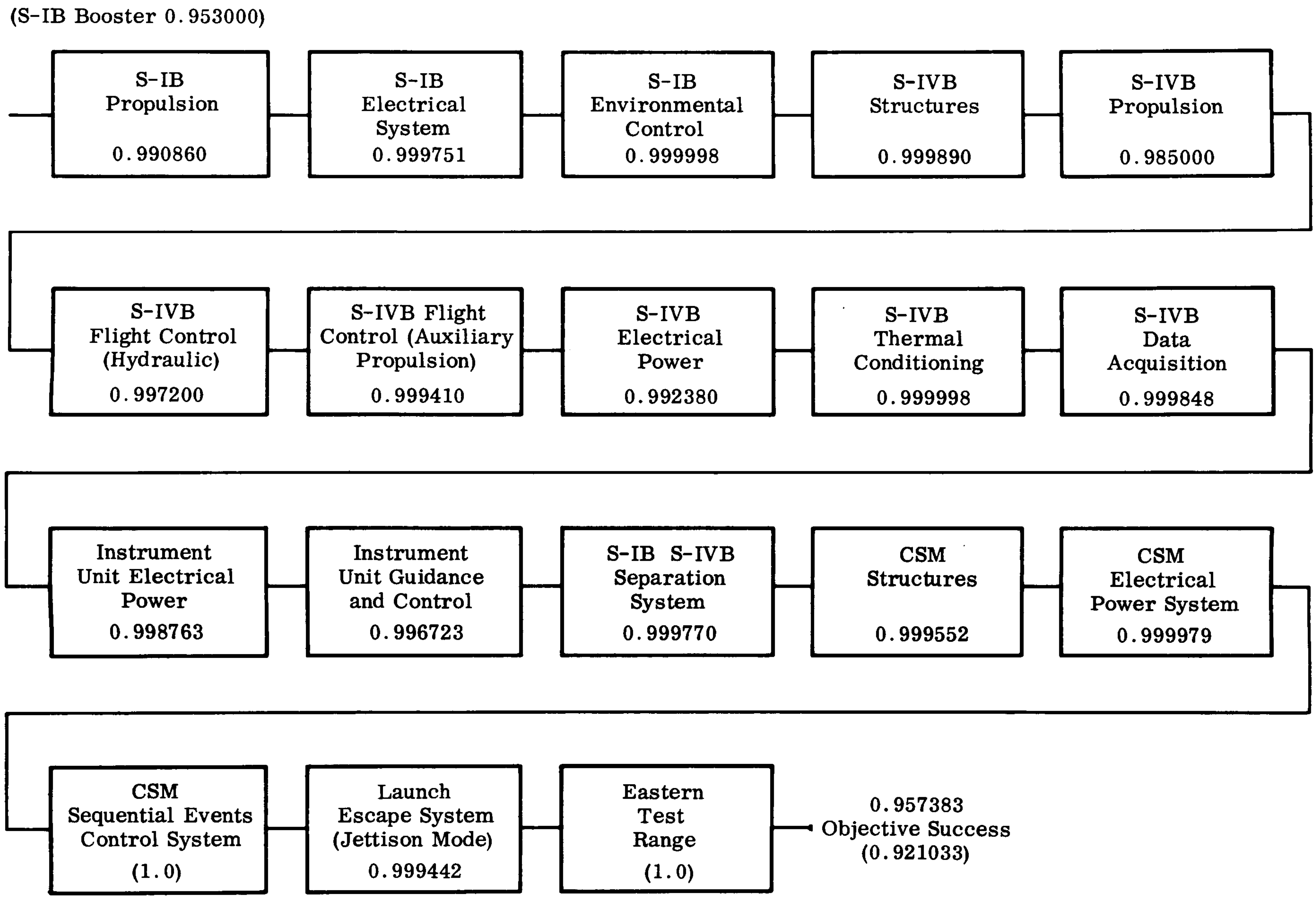


Figure A-2(d). Demonstration of Launch Vehicle Structural Integrity

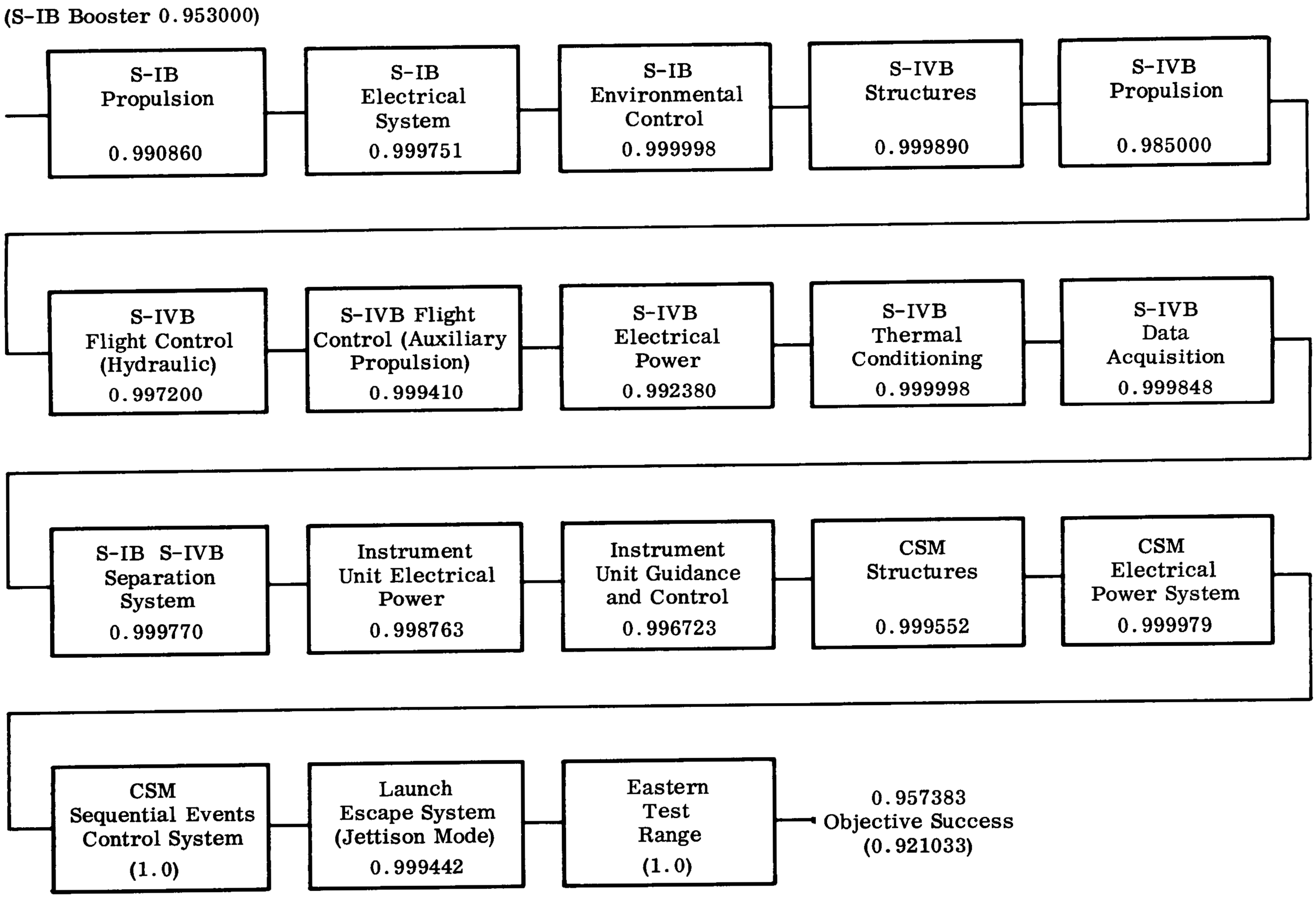


Figure A-2(e). Verification of Launch Vehicle Propulsion System Operation

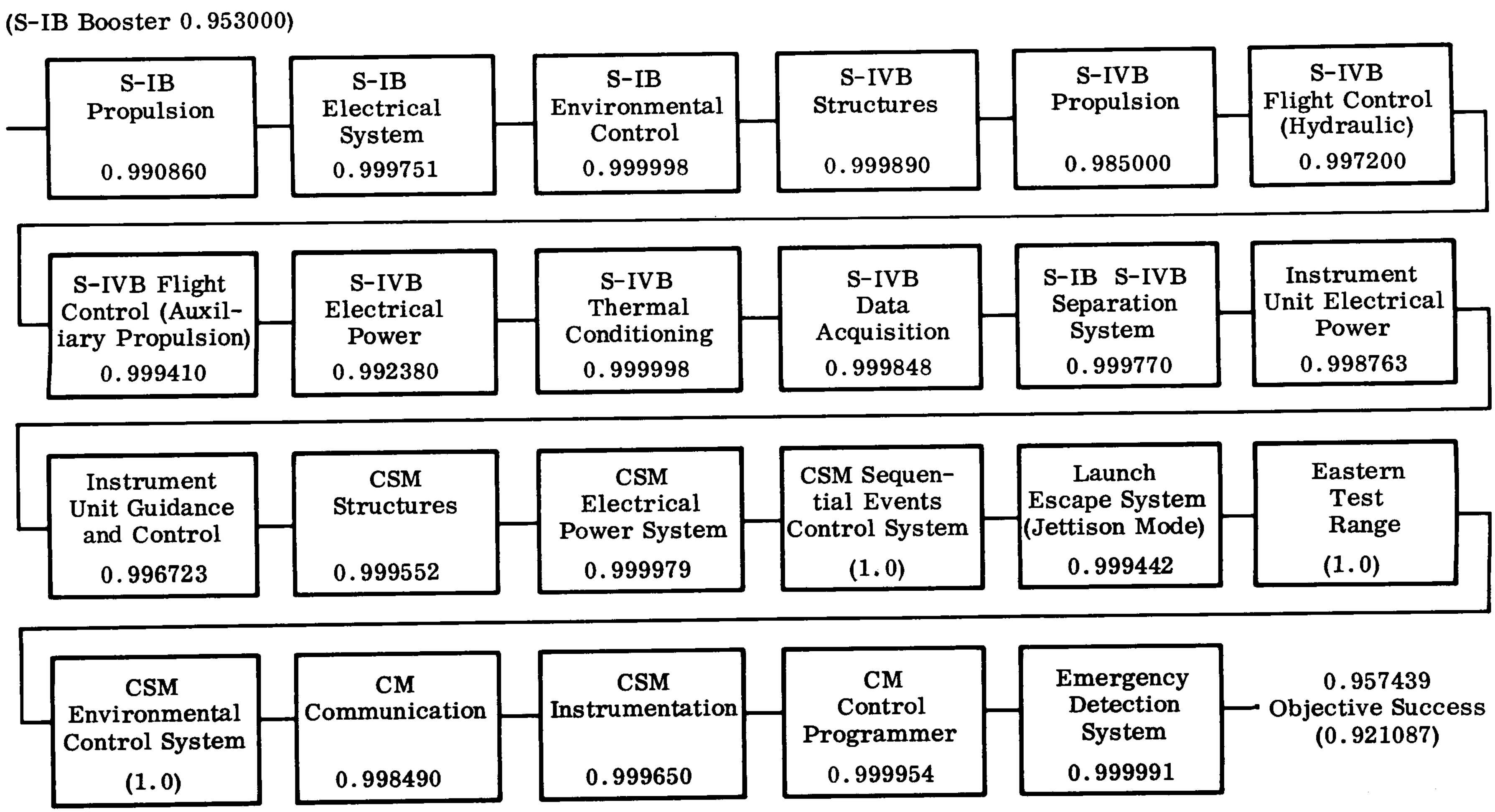


Figure A-2(f). Evaluation of Performance of Open-Loop Malfunction (Emergency) Detection System

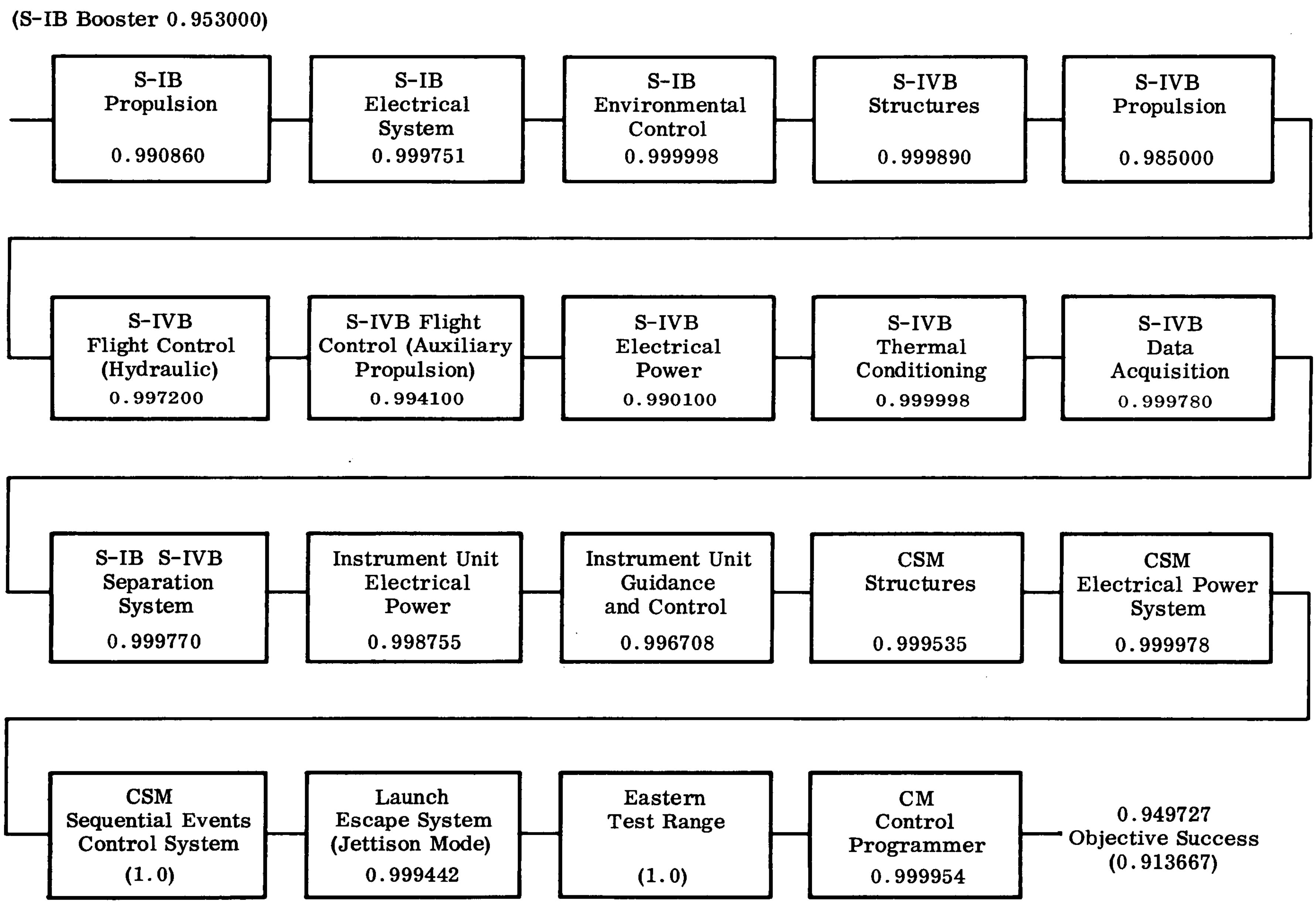


Figure A-2(g). Verification of Launch Vehicle Guidance and Control Subsystem Operation

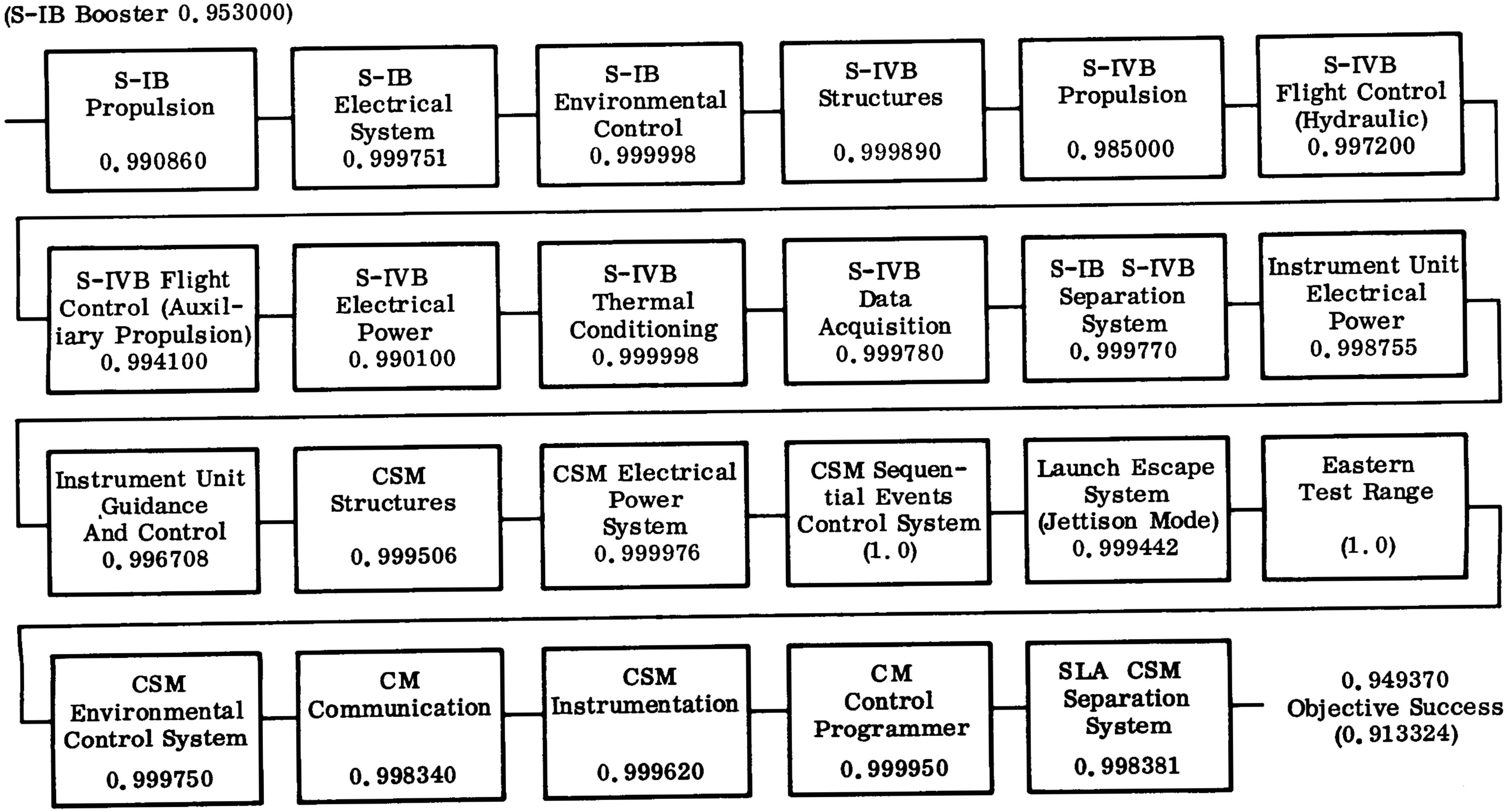


Figure A-2(h). Demonstration of Launch Vehicle CSM Separation

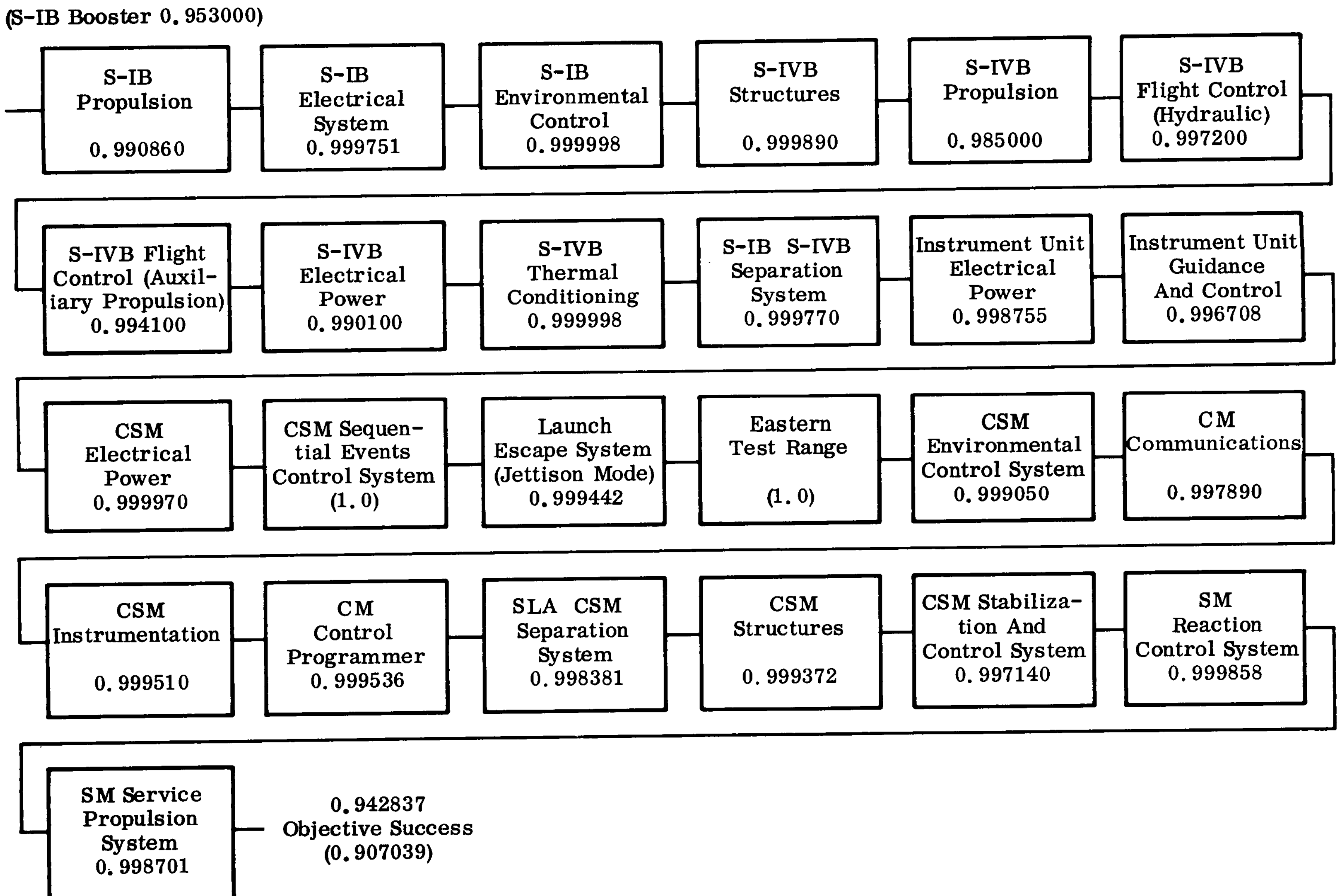


Figure A-2(i). Determination of Long-Duration Service Propulsion System Performance Including Shutdown

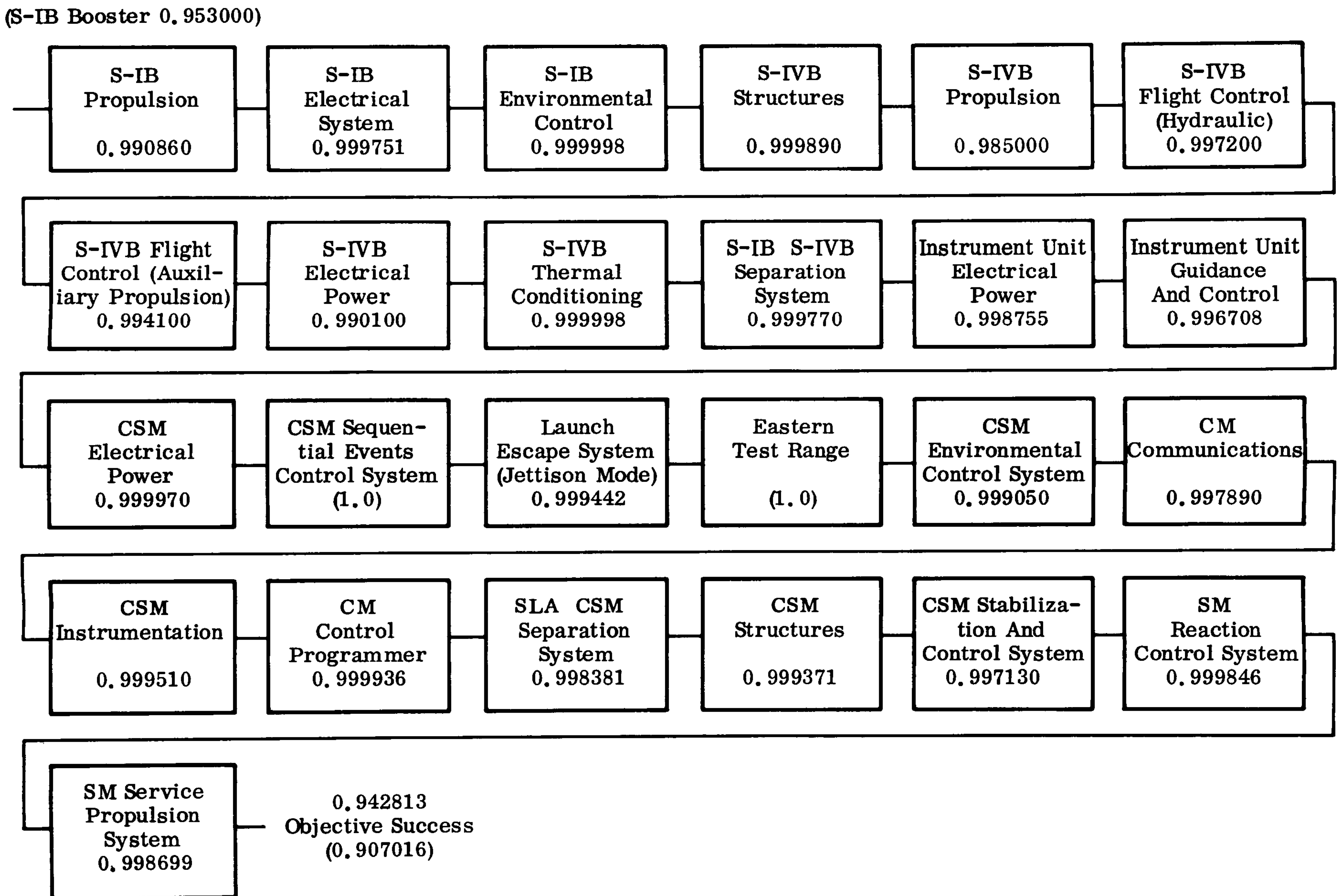


Figure A-2(j). Demonstration of Restart of Service Propulsion System Following Long-Duration Burn

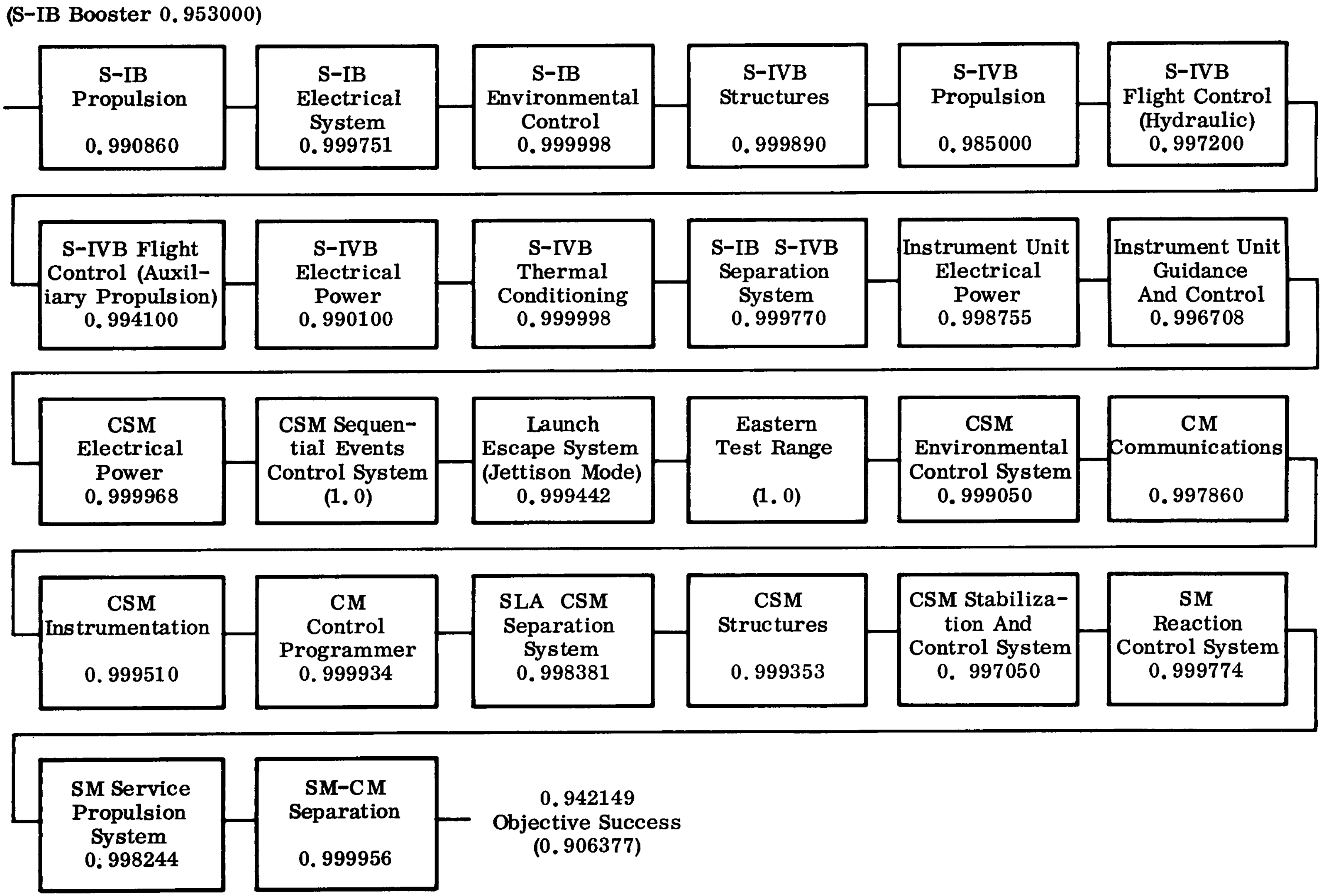


Figure A-2(k). Verification of Spacecraft SM RCS Subsystem Operation

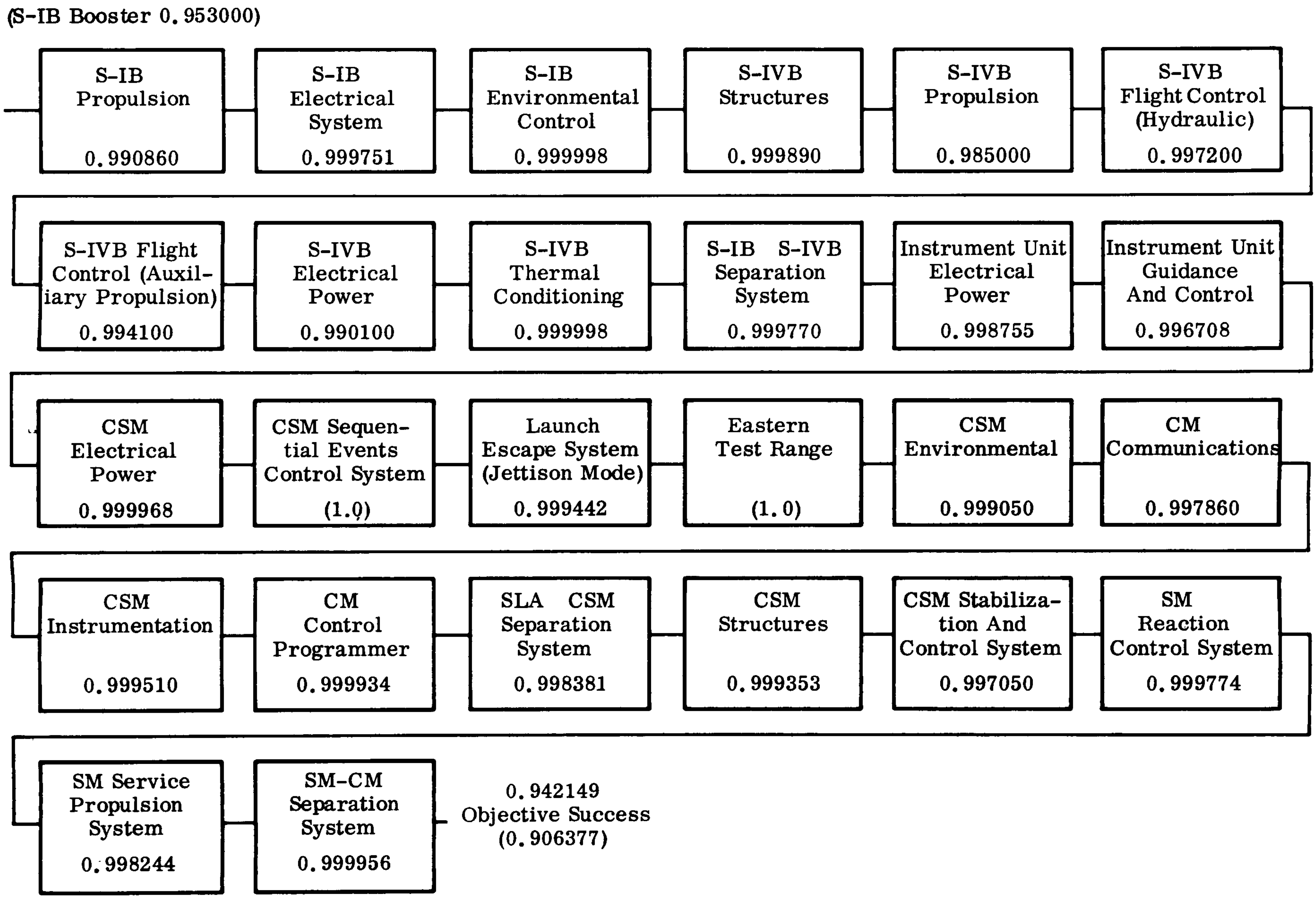


Figure A-2(l). Demonstration of SM CM Separation

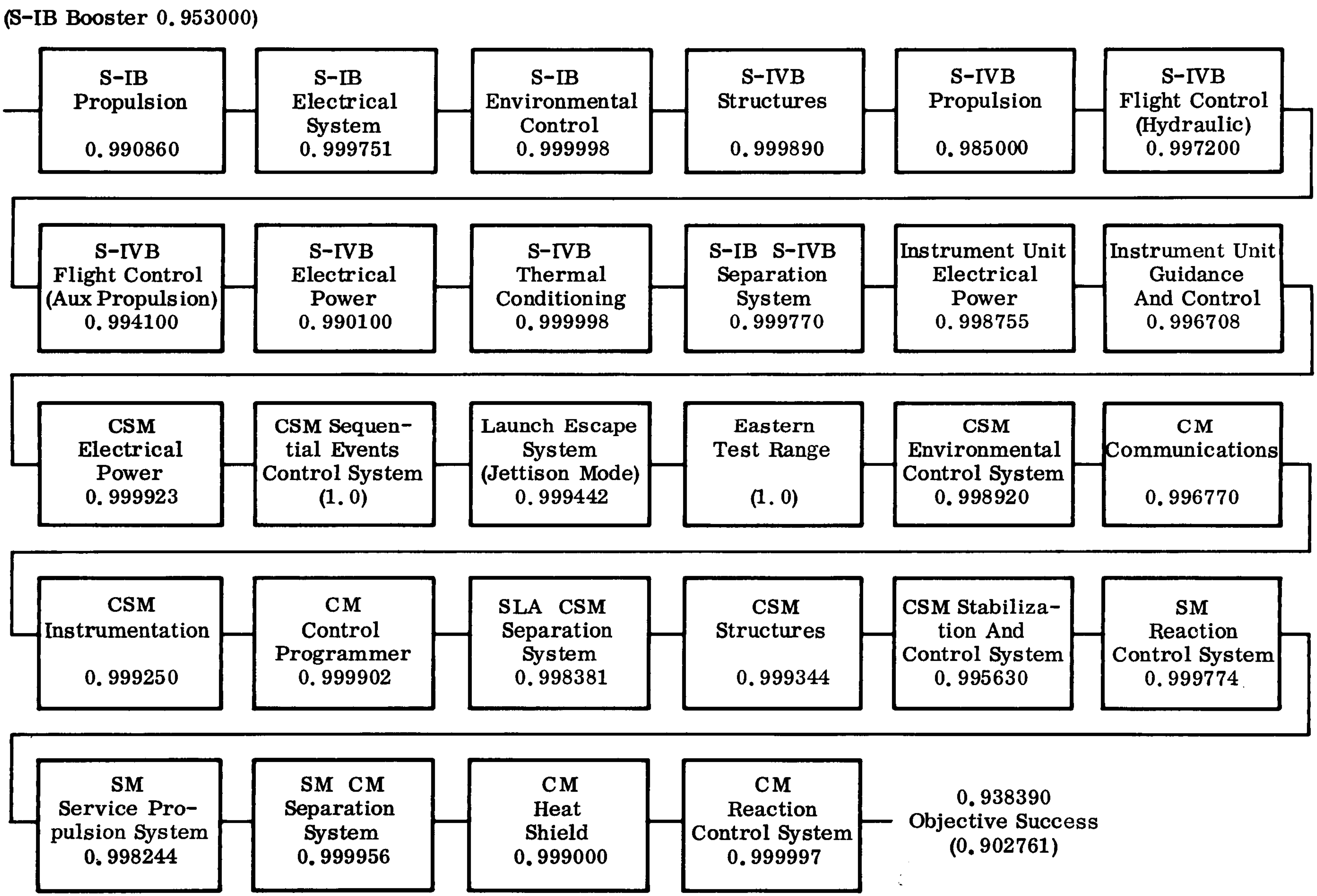


Figure A-2(m). Evaluation of CM Heat Shield Performance

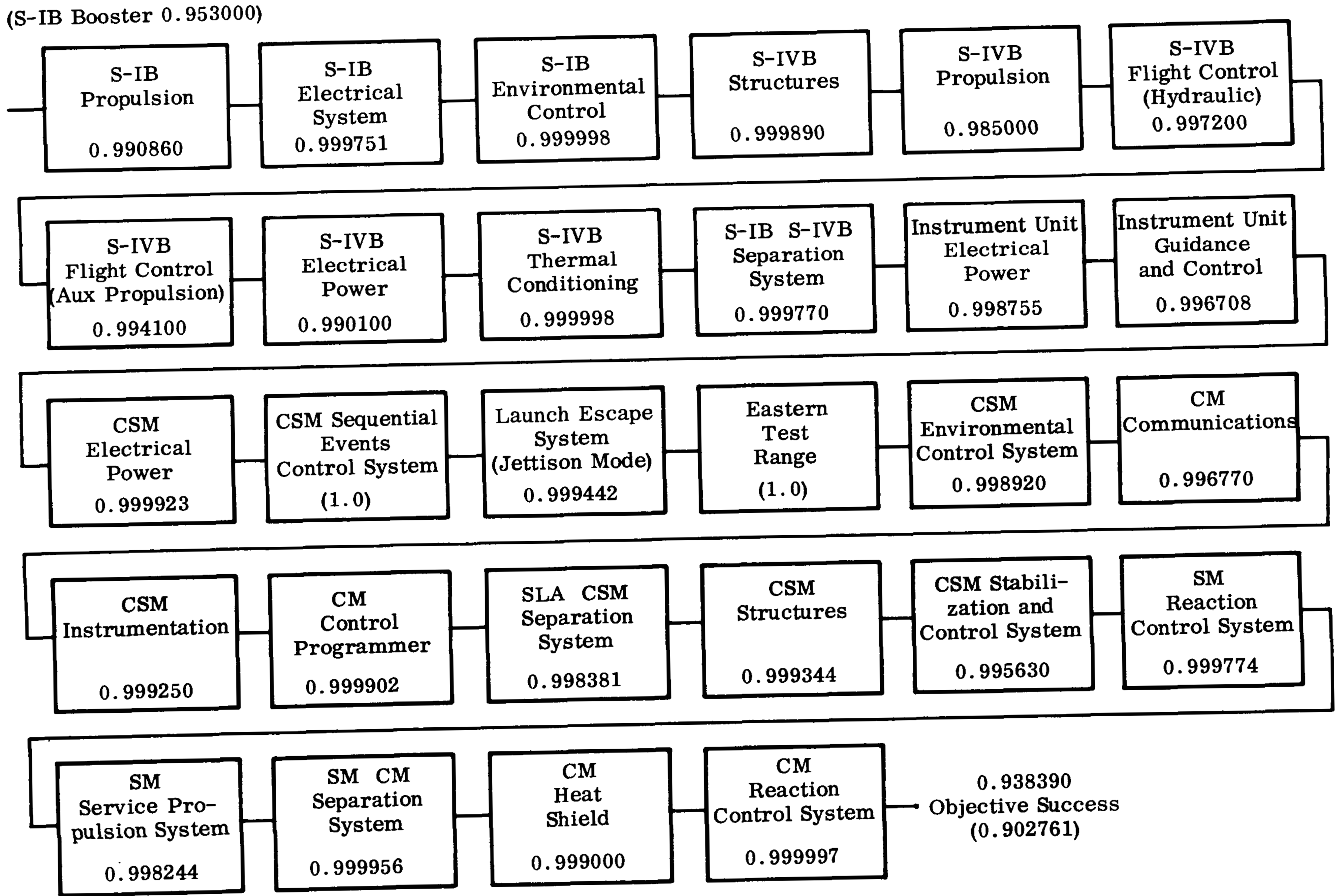


Figure A-2(n). Verification of Spacecraft Stabilization and Control System Operation

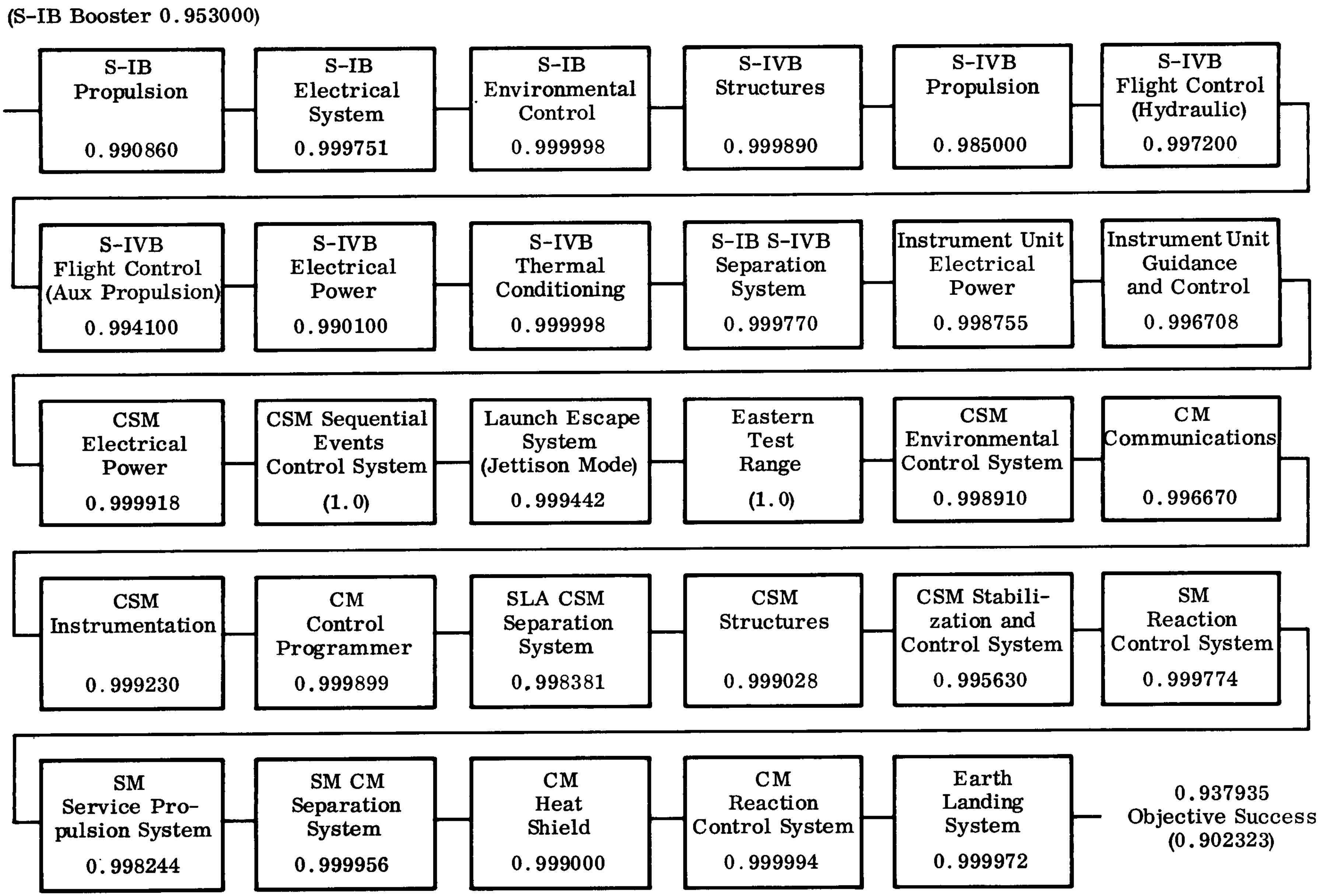


Figure A-2(o). Verification of Spacecraft CM Reaction Control System Operation

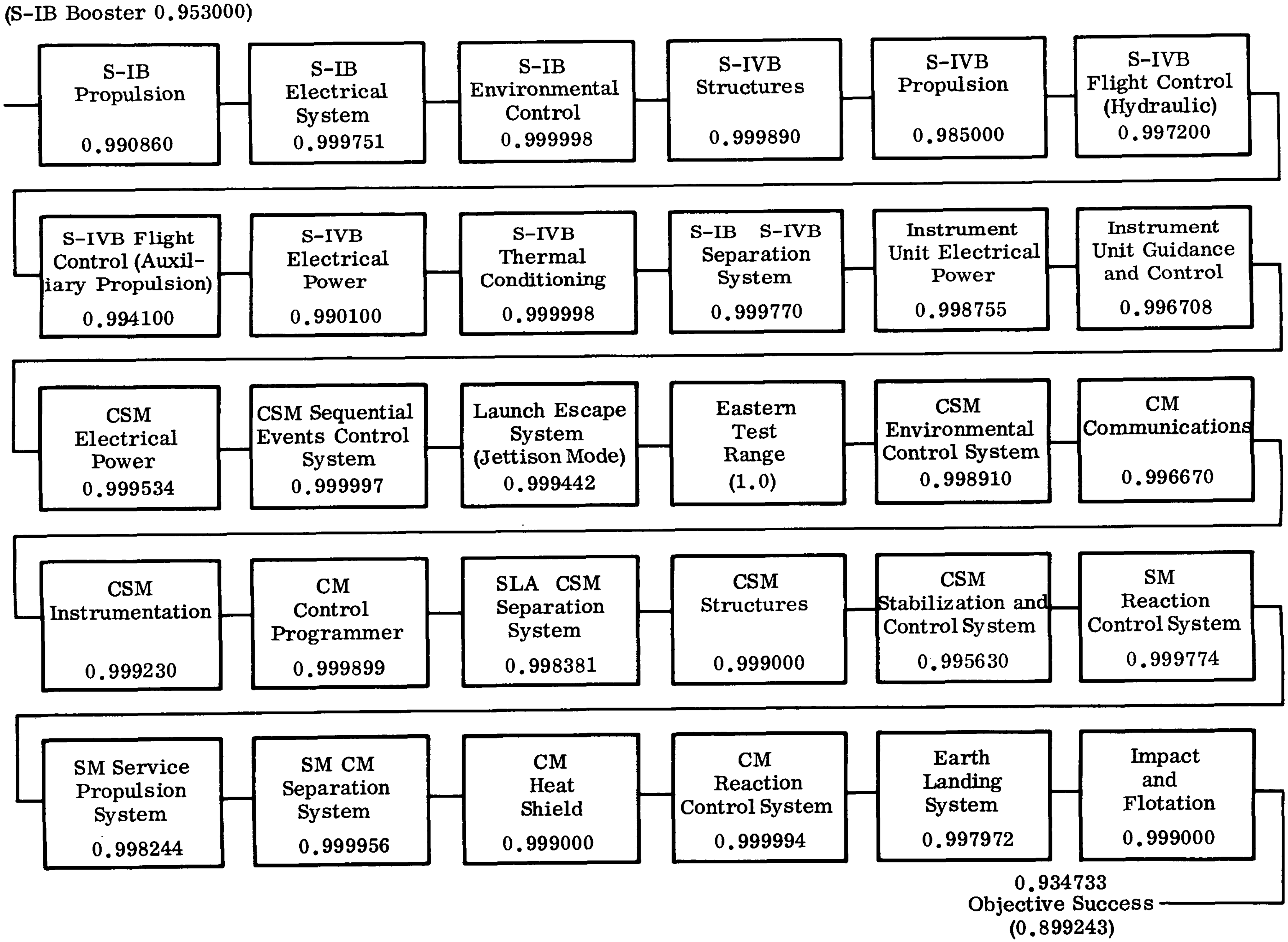


Figure A-2(p). Verification of Spacecraft Communication System Operation

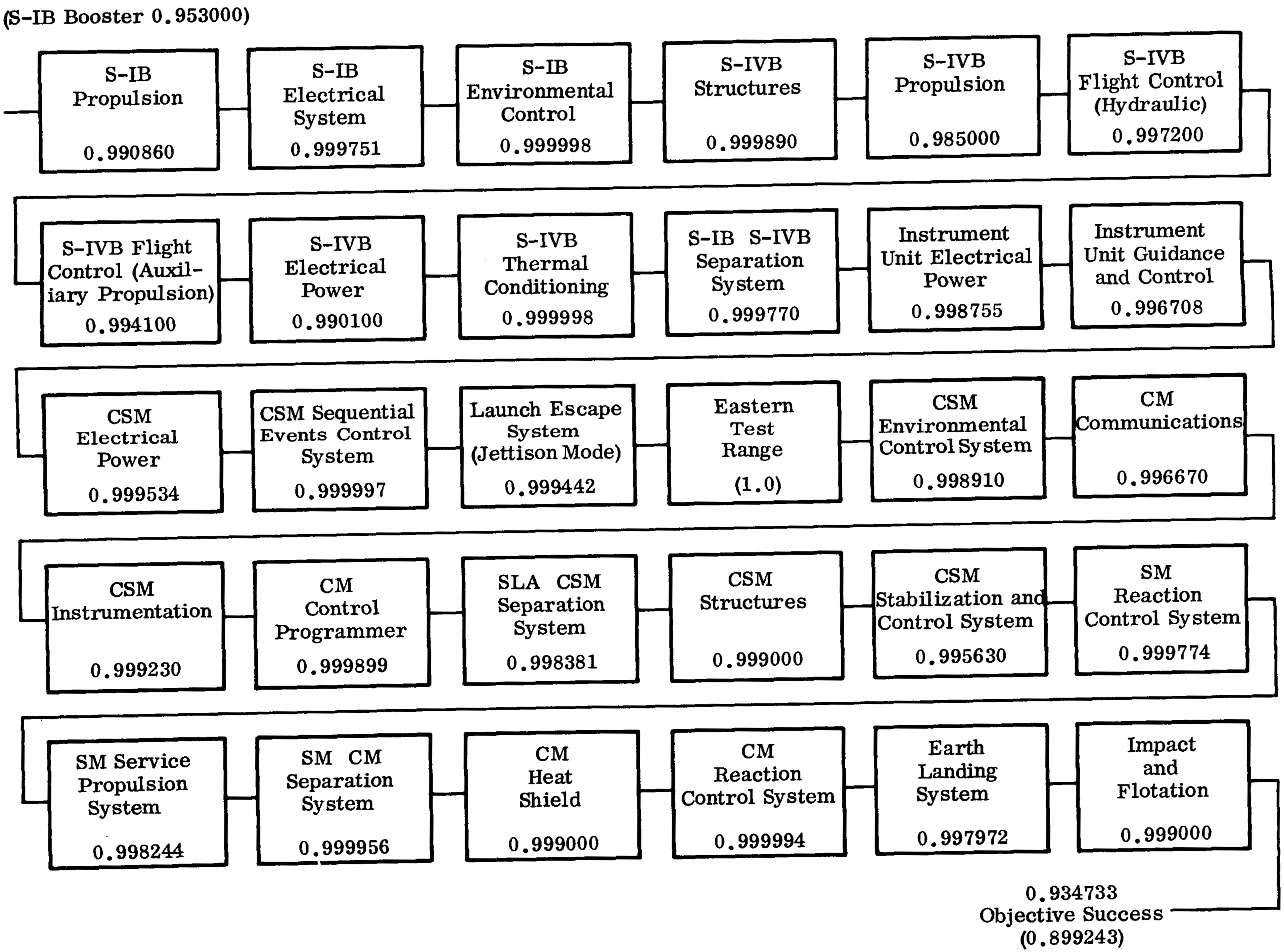


Figure A-2(q). Verification of Spacecraft Earth Landing System Operation

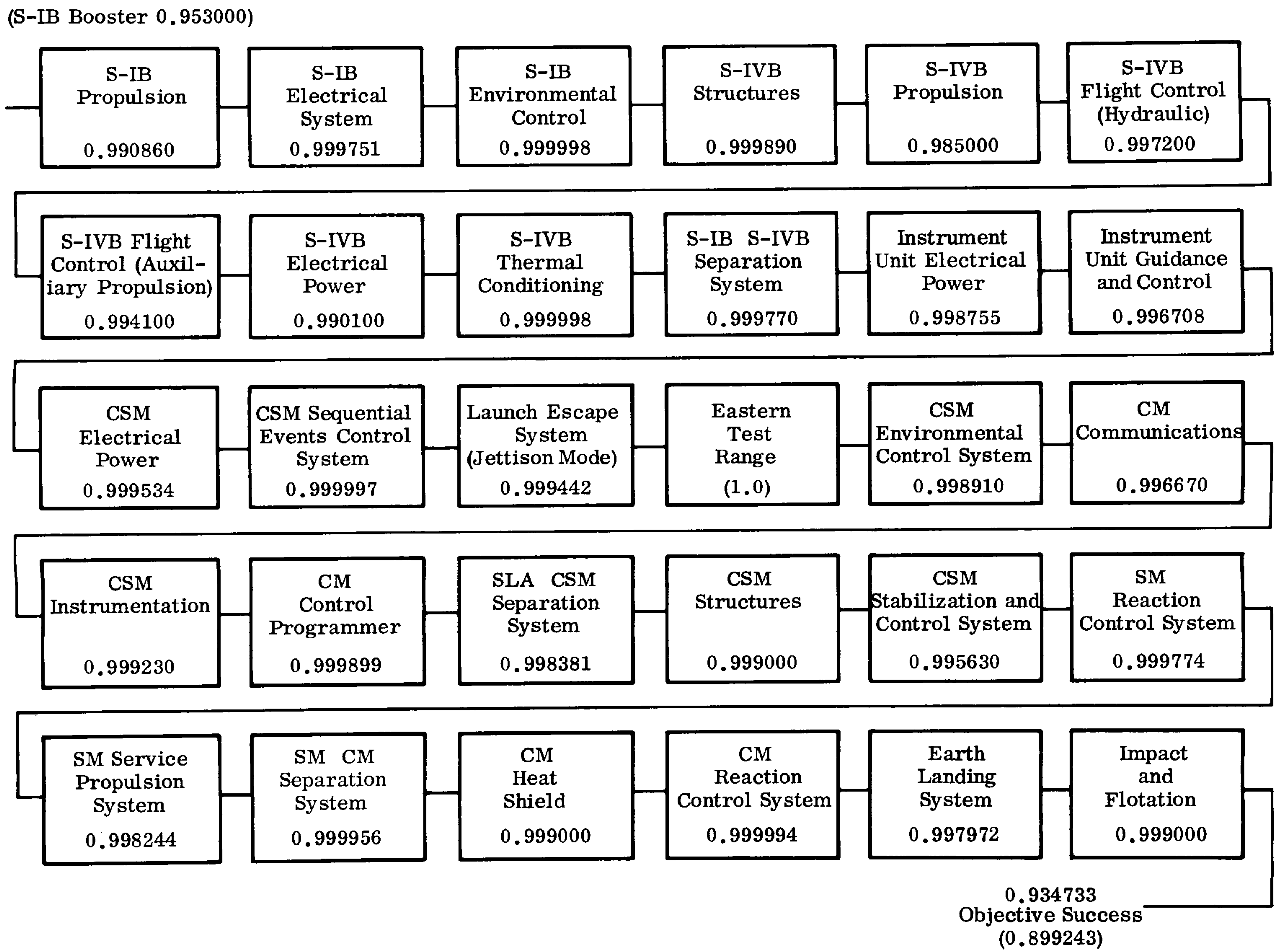


Figure A-2(r). Verification of Spacecraft Environmental Control System Operation

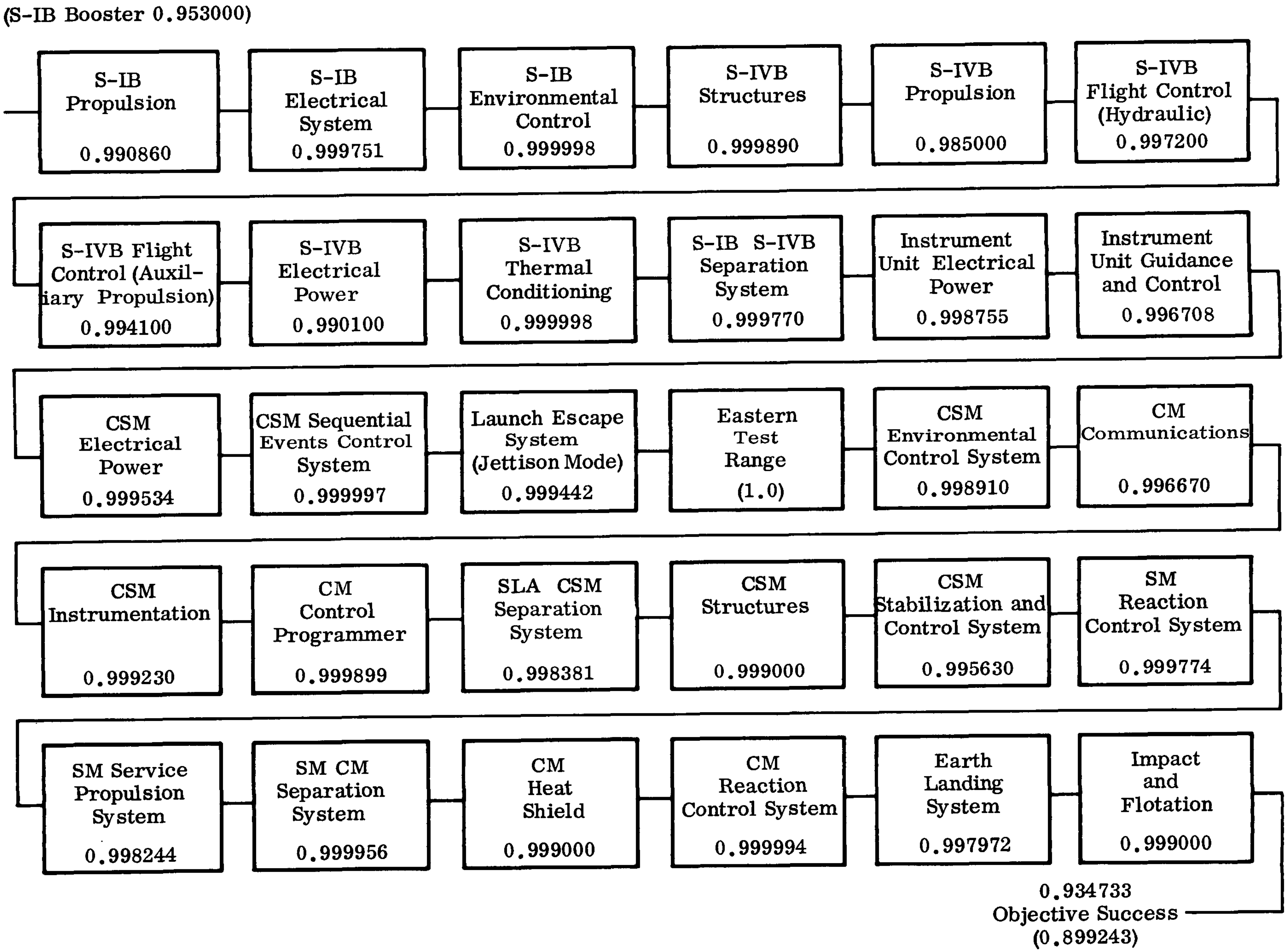


Figure A-2(s). Verification of Spacecraft Electrical Power System Operation

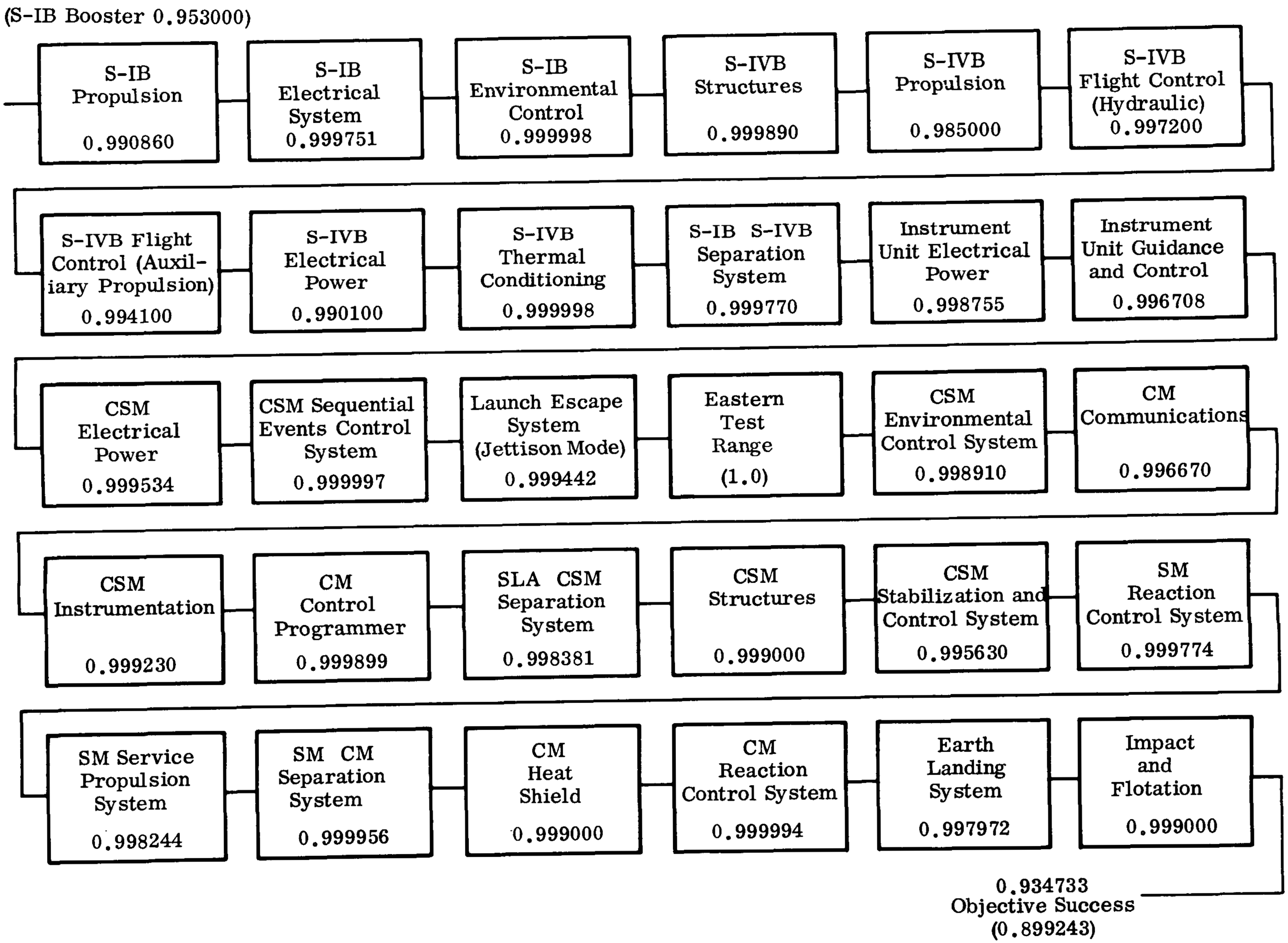


Figure A-2(t). Determination of Adequacy of Recovery Aids

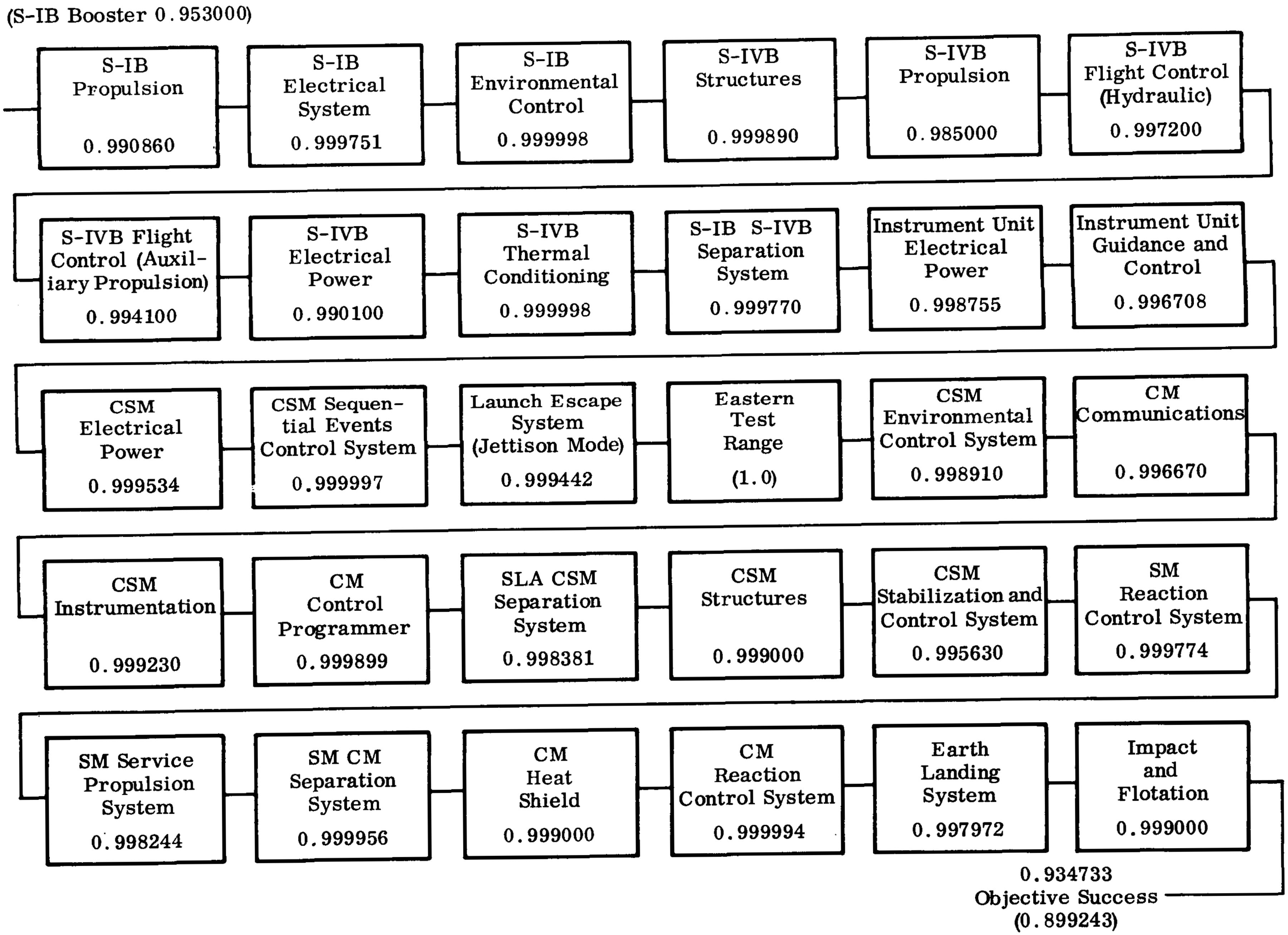


Figure A-2(u). Determination of CM Adequacy for Entry from Low Earth Orbit

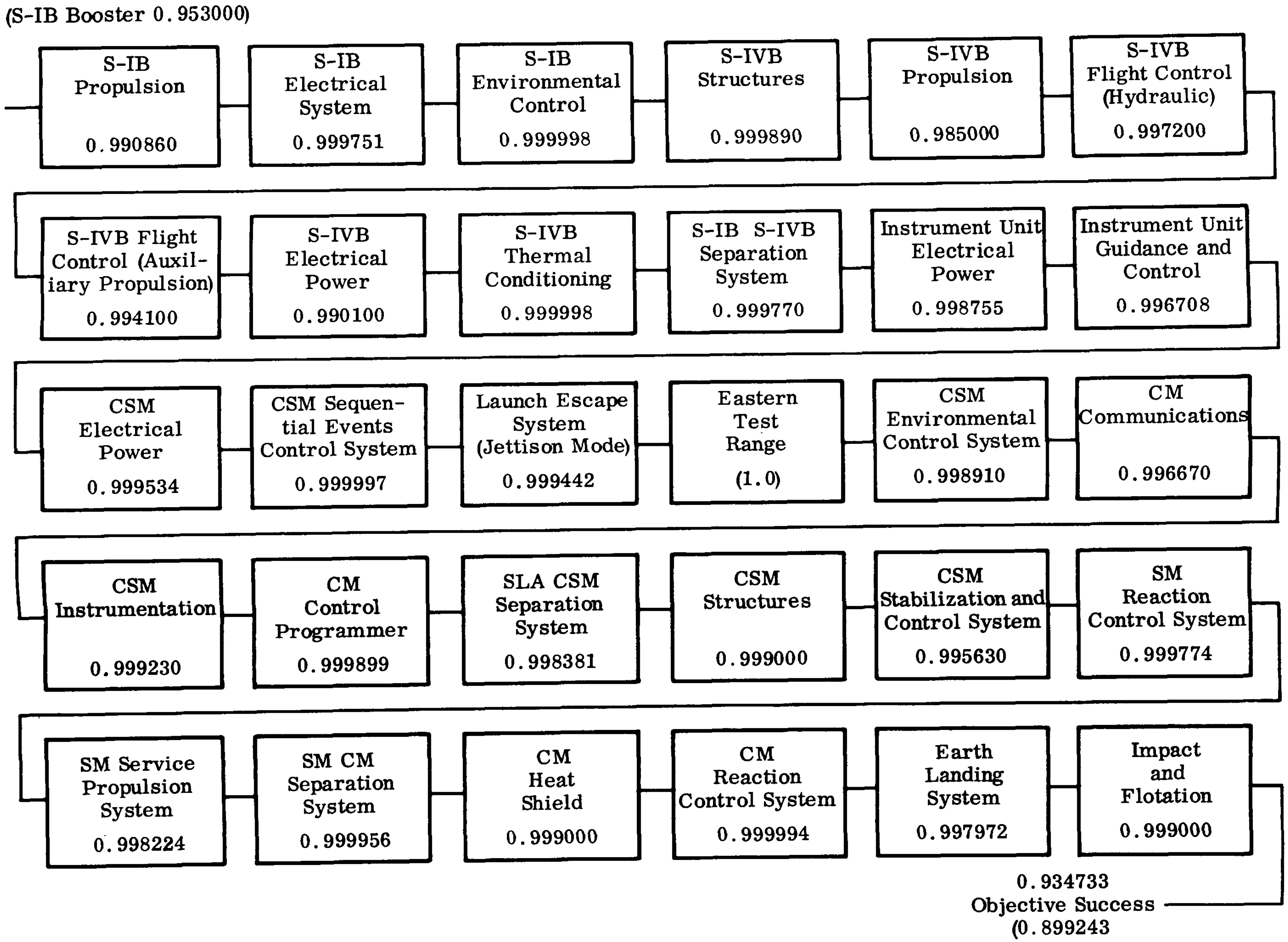


Figure A-2(v). Demonstration of Spacecraft Structural Integrity

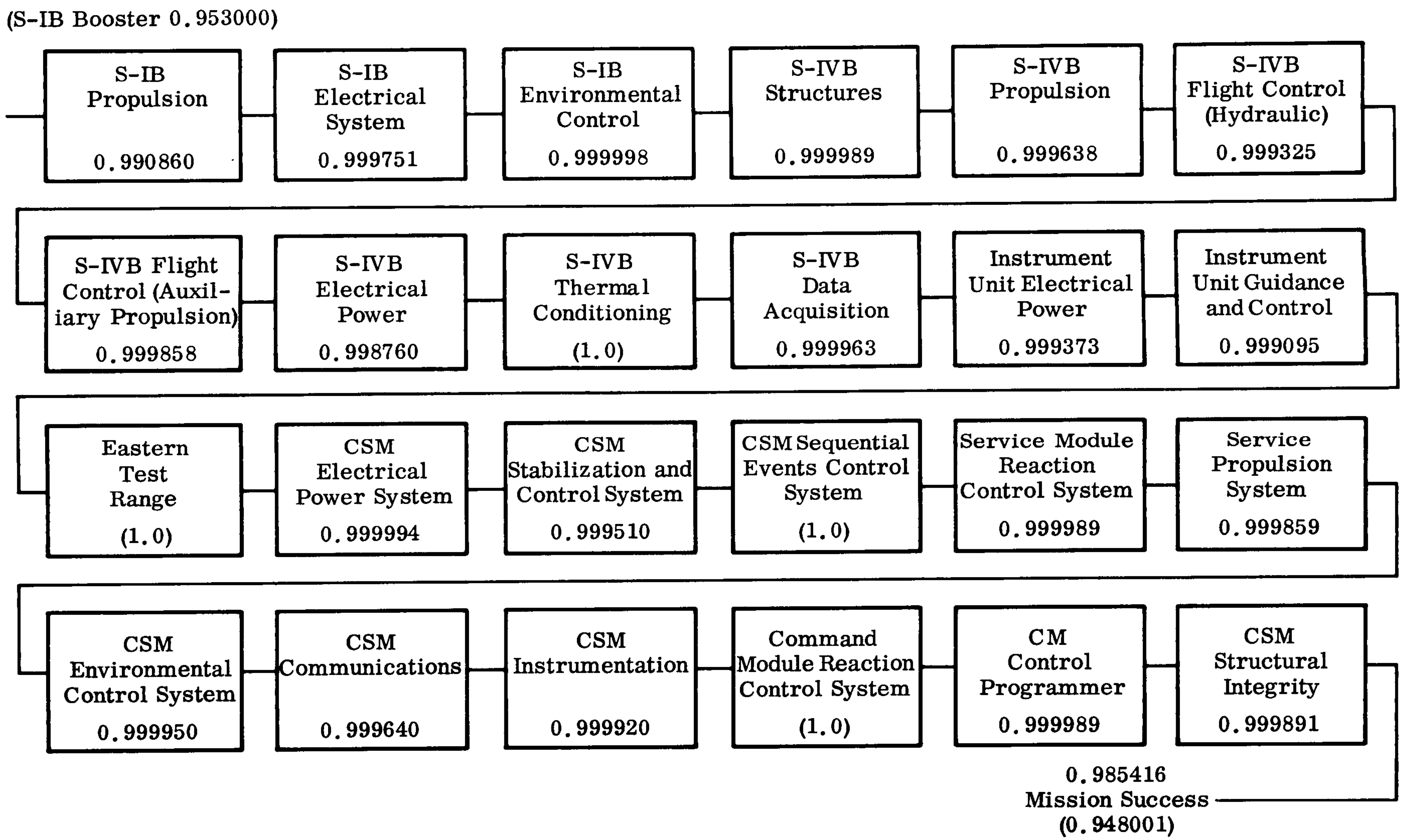


Figure A-3(a). Prediction Model for Mission Success, Liftoff Through S-IB Cutoff, Phase 2

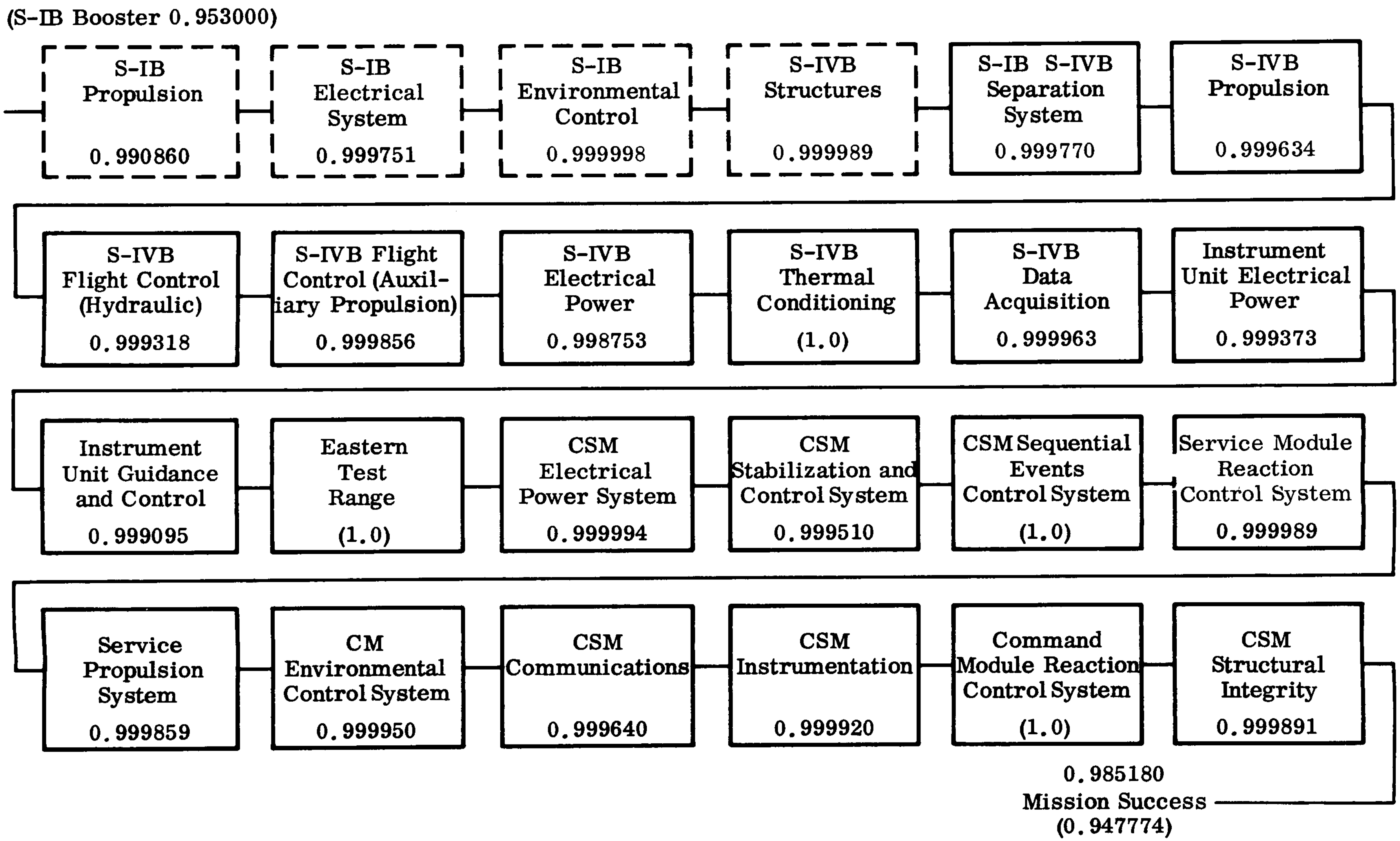


Figure A-3(b). Prediction Model for Mission Success, S-IB Cutoff Through S-IB Separation from S-IVB, Phase 3

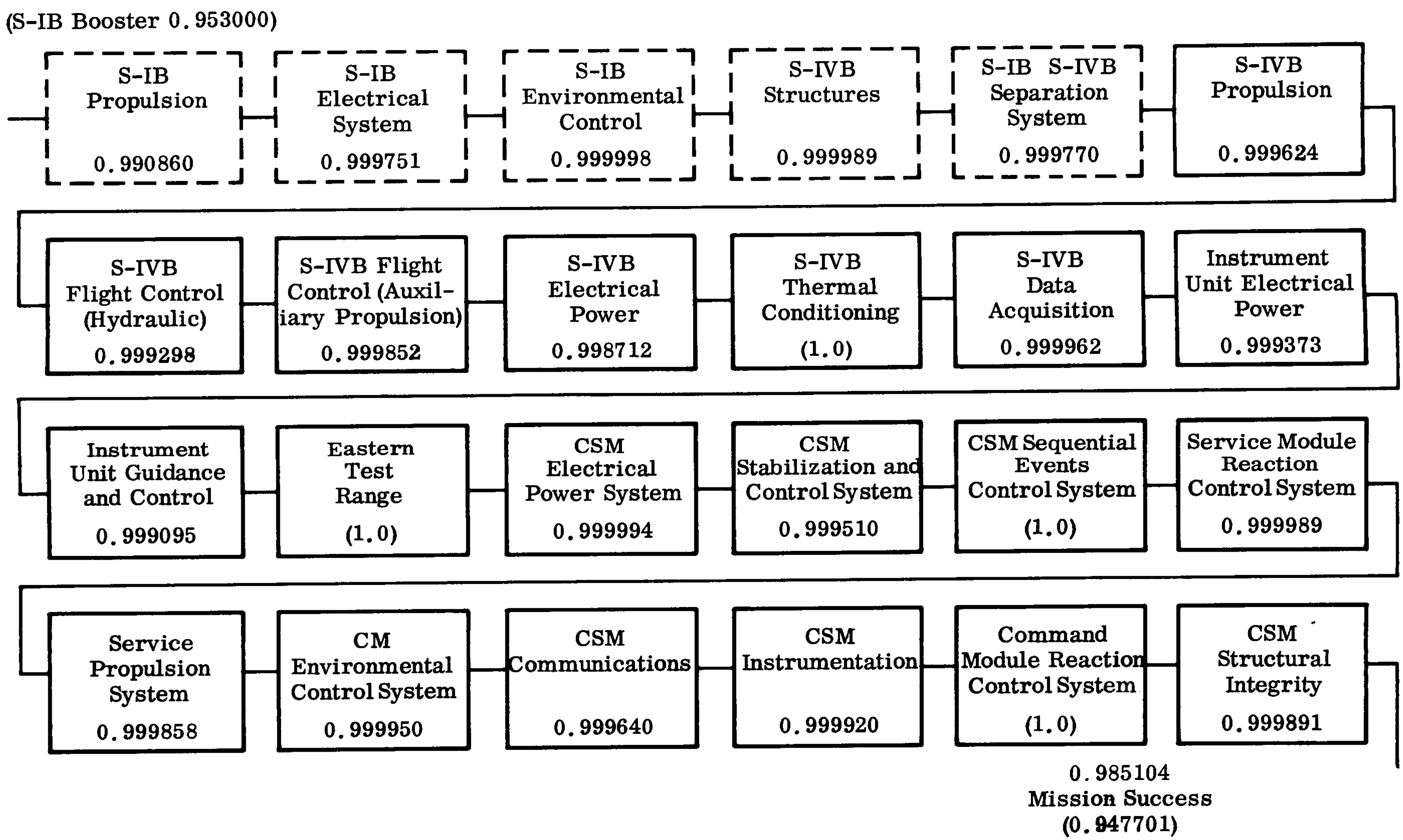


Figure A-3(c). Prediction Model for Mission Success, S-IB Separation to S-IVB Ignition, Phase 4

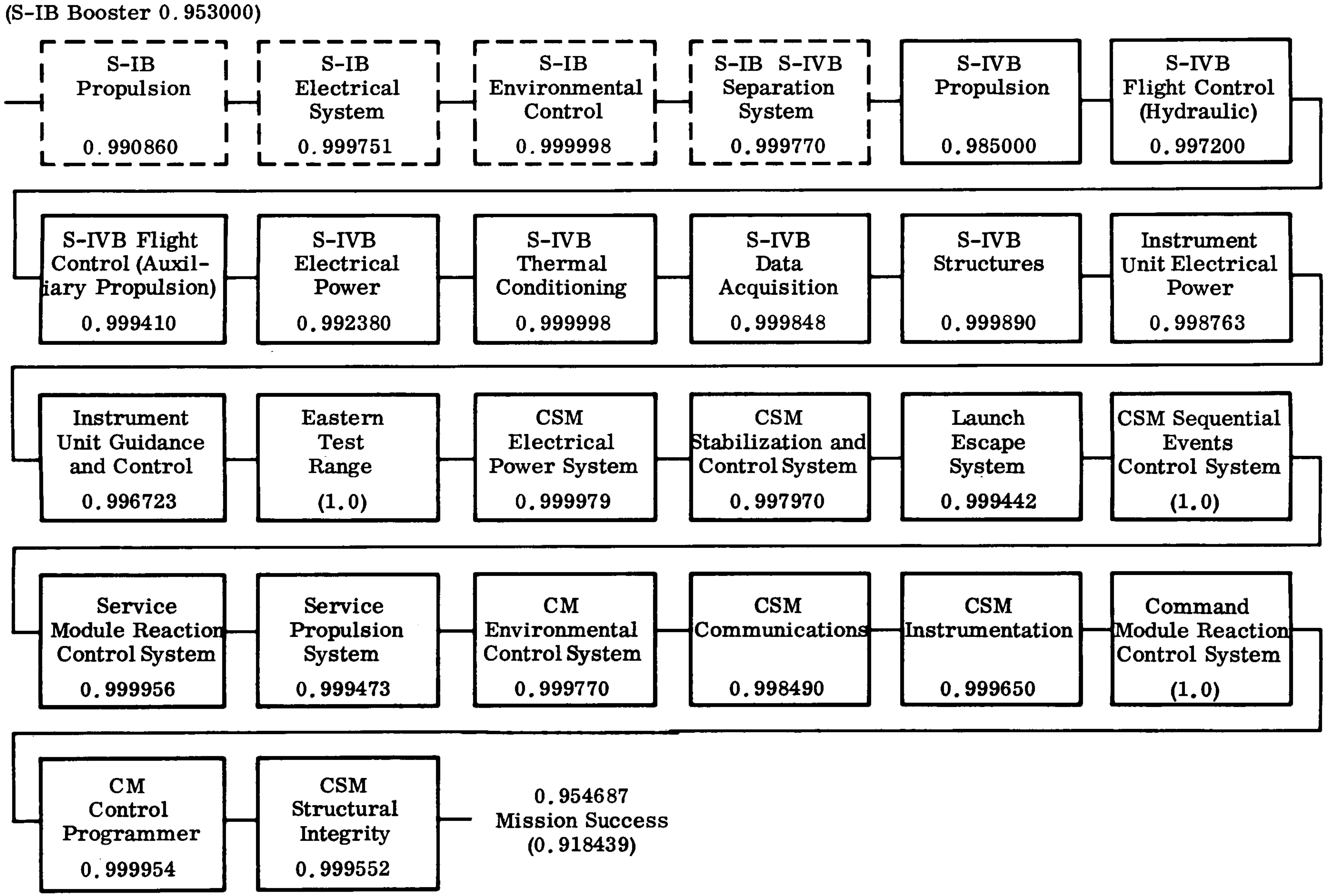


Figure A-3(d). Prediction Model for Mission Success, S-IVB Ignition Through Cutoff, Phase 5

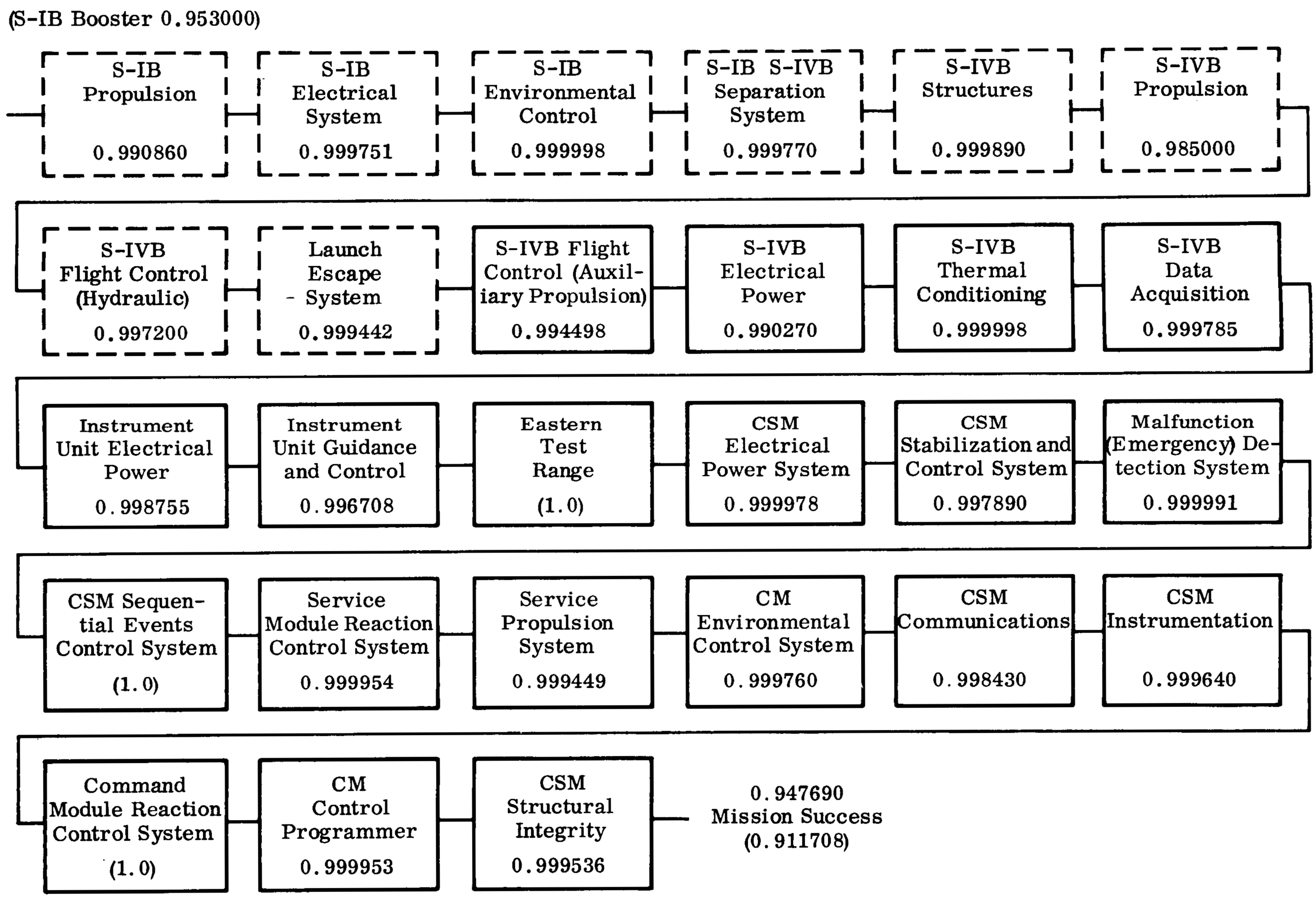


Figure A-3(e). Prediction Model for Mission Success,
S-IVB Engine Cutoff through Coast and Orientation Maneuver, Phase 6

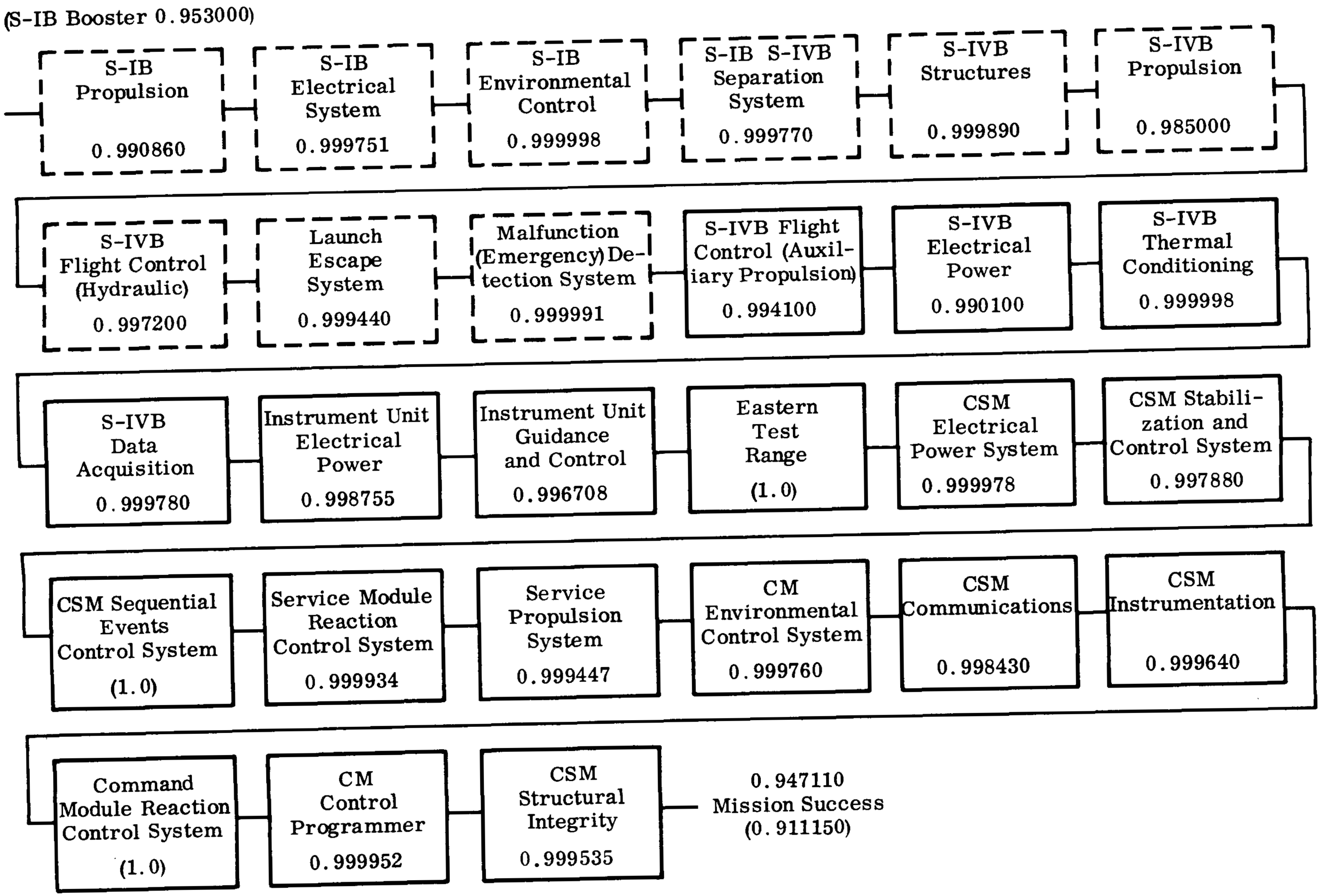


Figure A-3(f). Prediction Model for Mission Success, End Orientation Maneuver to S-IVB CSM Separation, Phase 7

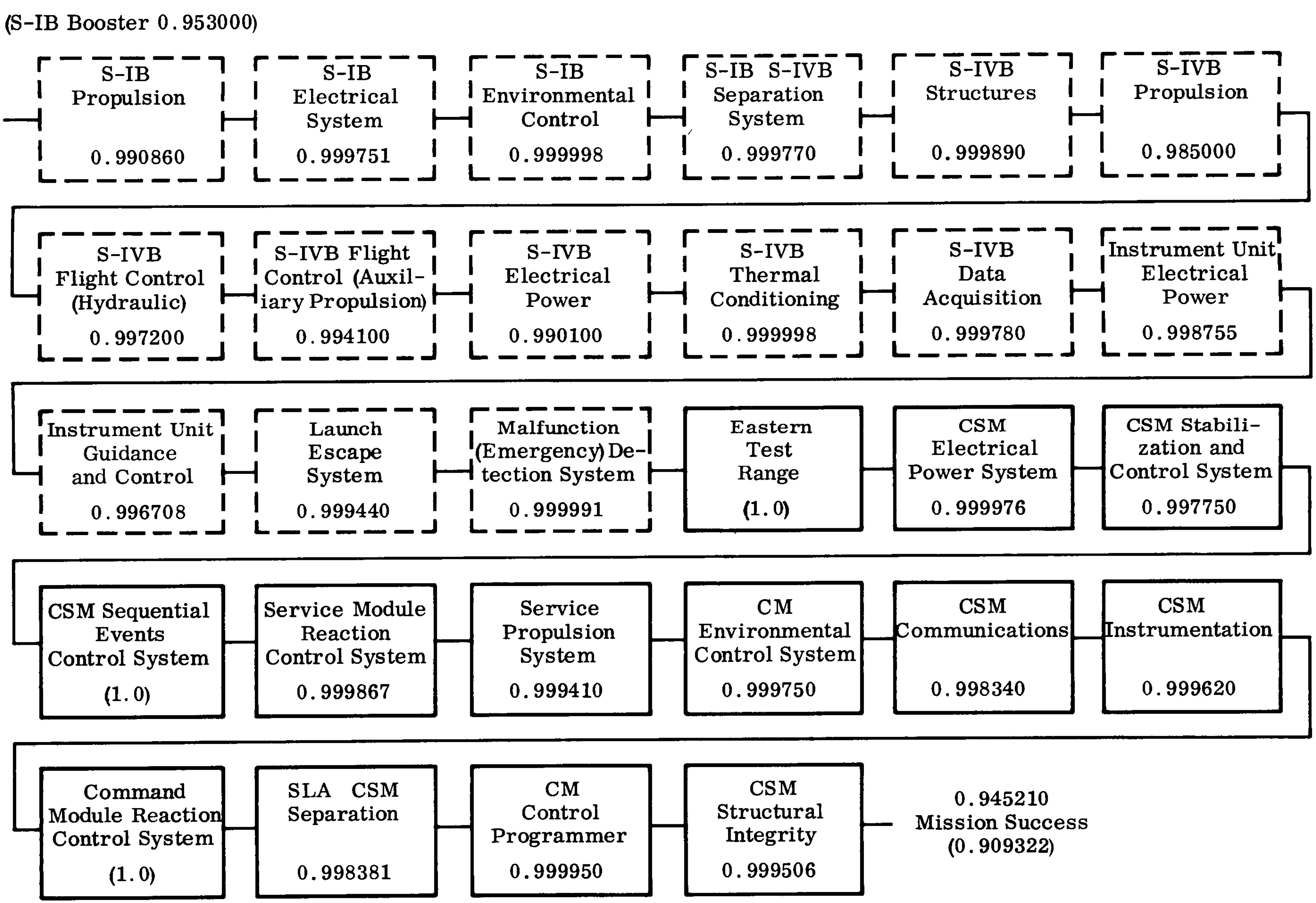


Figure A-3(g). Prediction Model for Mission Success, S-IVB/IU/SLA CSM Separation to SPS First Ignition, Phase 8

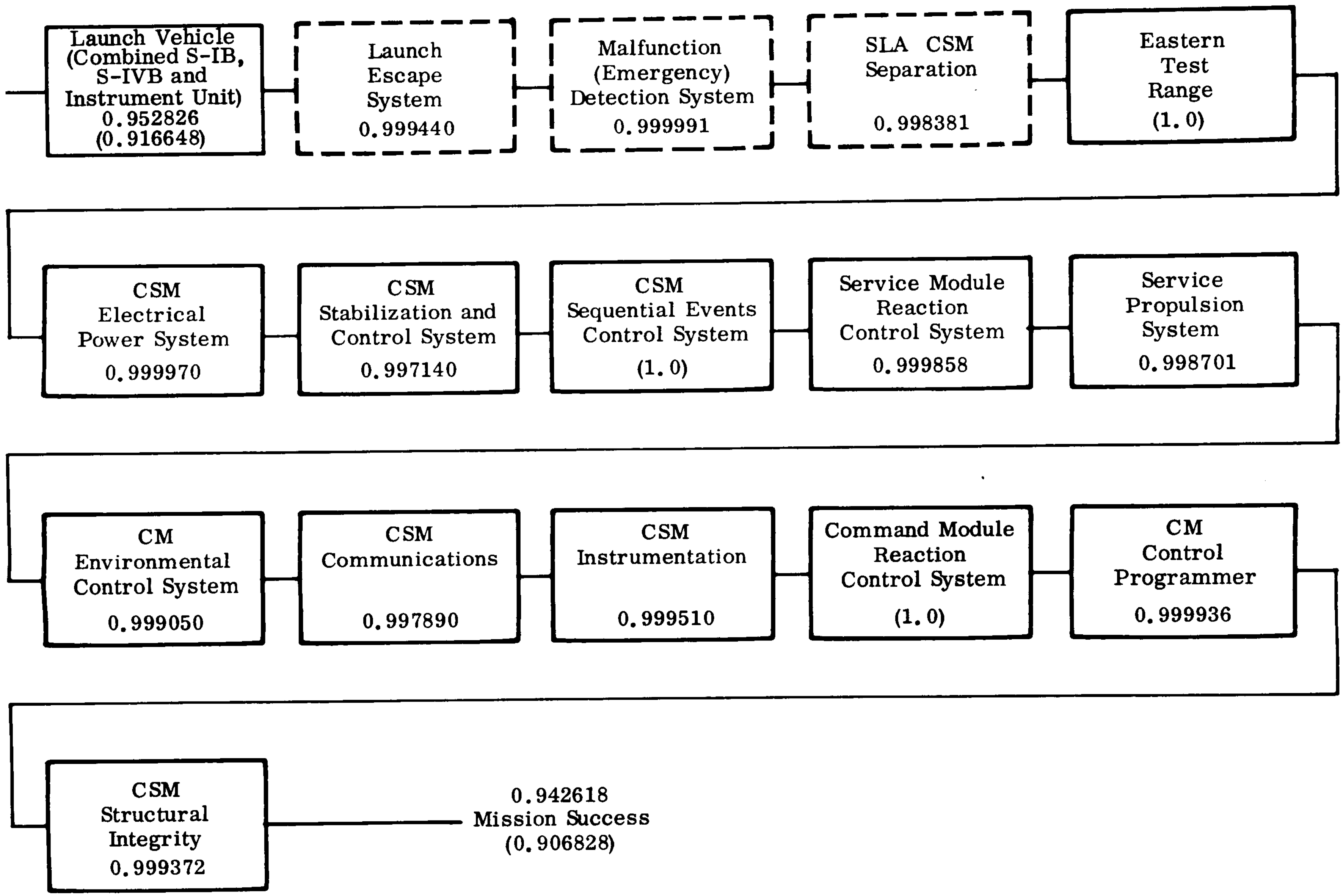


Figure A-3(h). Prediction Model for Mission Success, SPS First Ignition through SPS First Cutoff, Phase 9

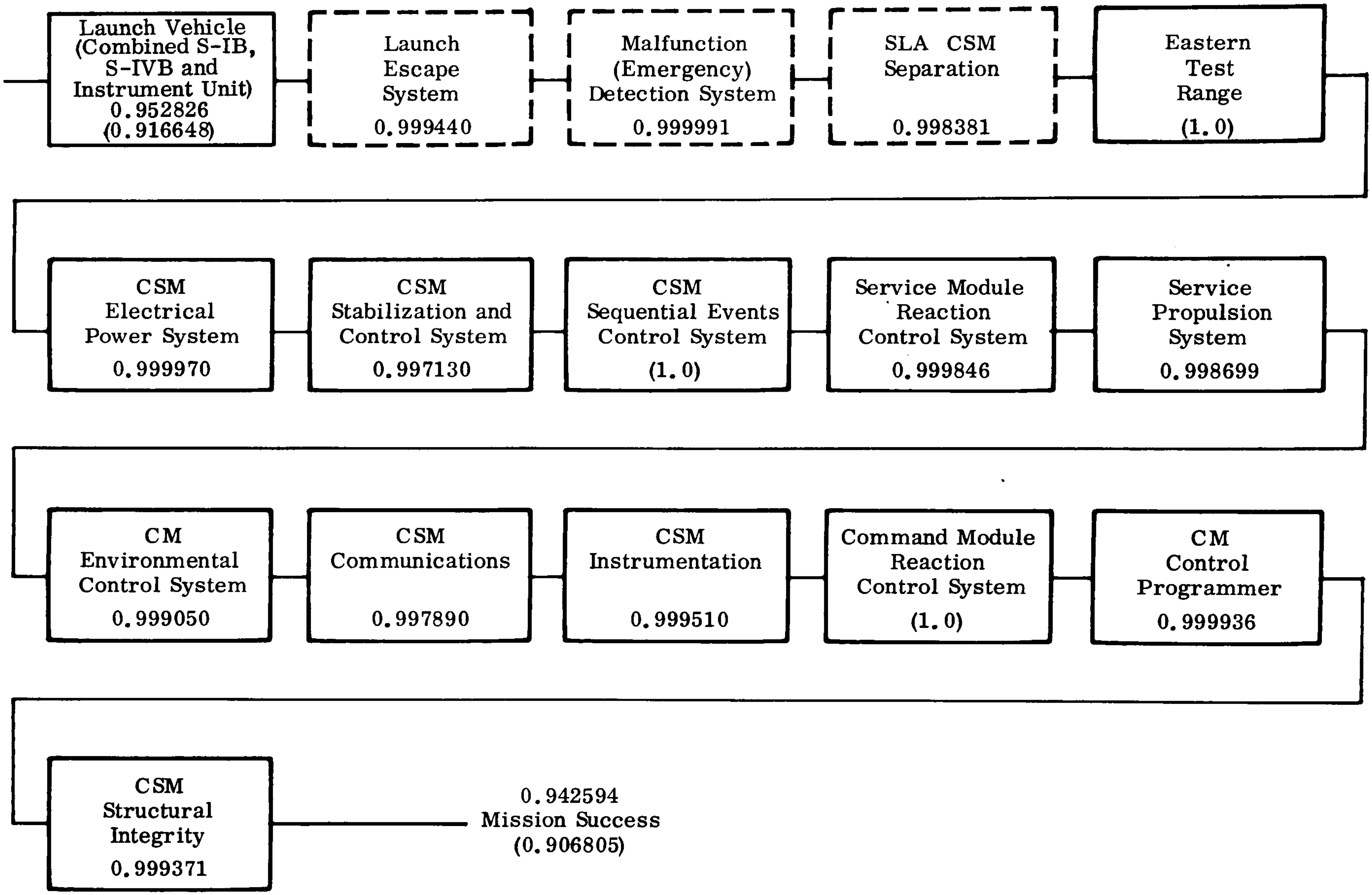


Figure A-3(i). Prediction Model for Mission Success, SPS First Cutoff to SPS Second Ignition, Phase 10

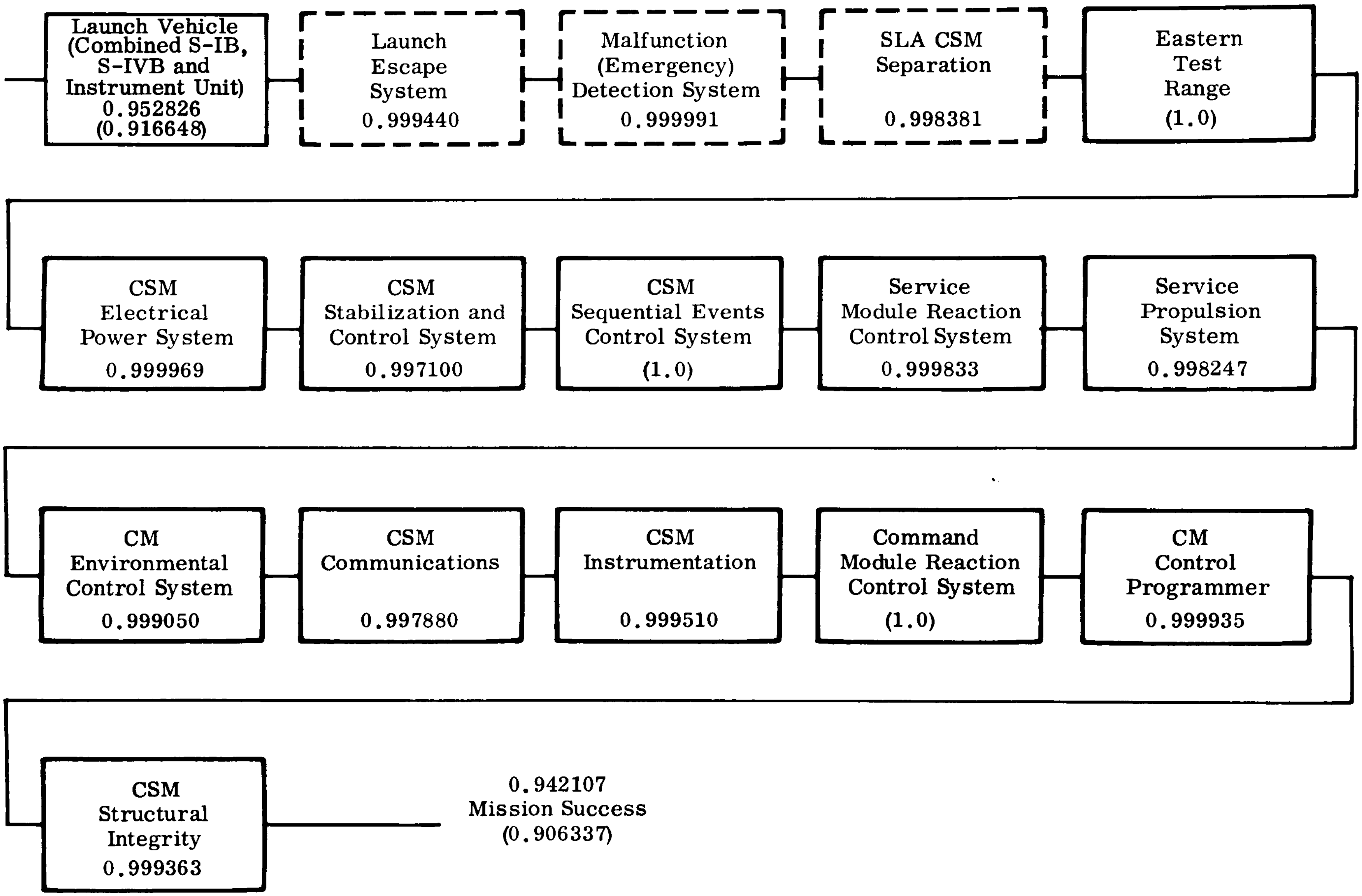


Figure A-3(j). Prediction Model for Mission Success, SPS Second Ignition through SPS Second Cutoff, Phase 11

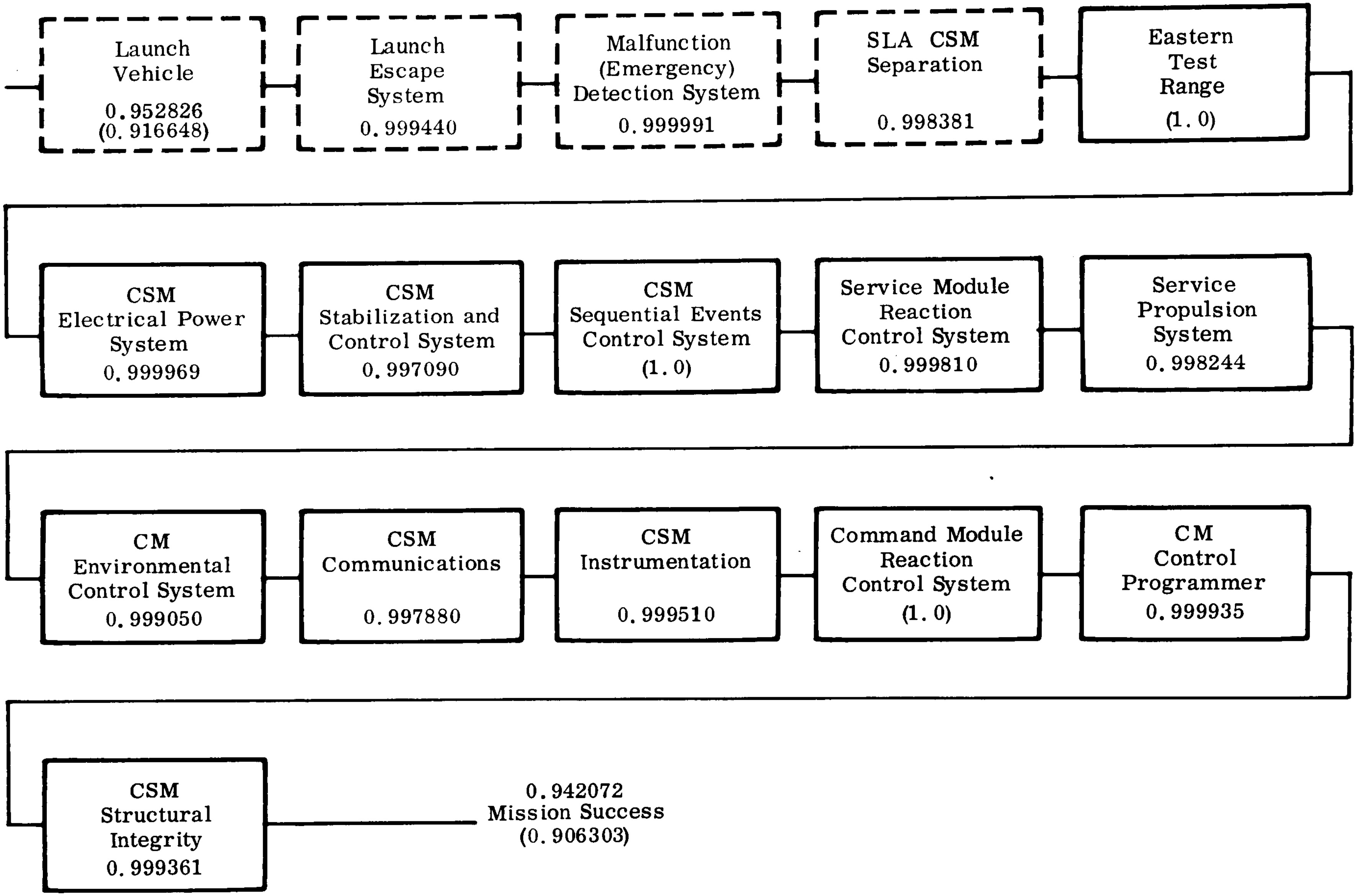


Figure A-3(k). Prediction Model for Mission Success, SPS Second Cutoff to SM CM Separation, Phase 12

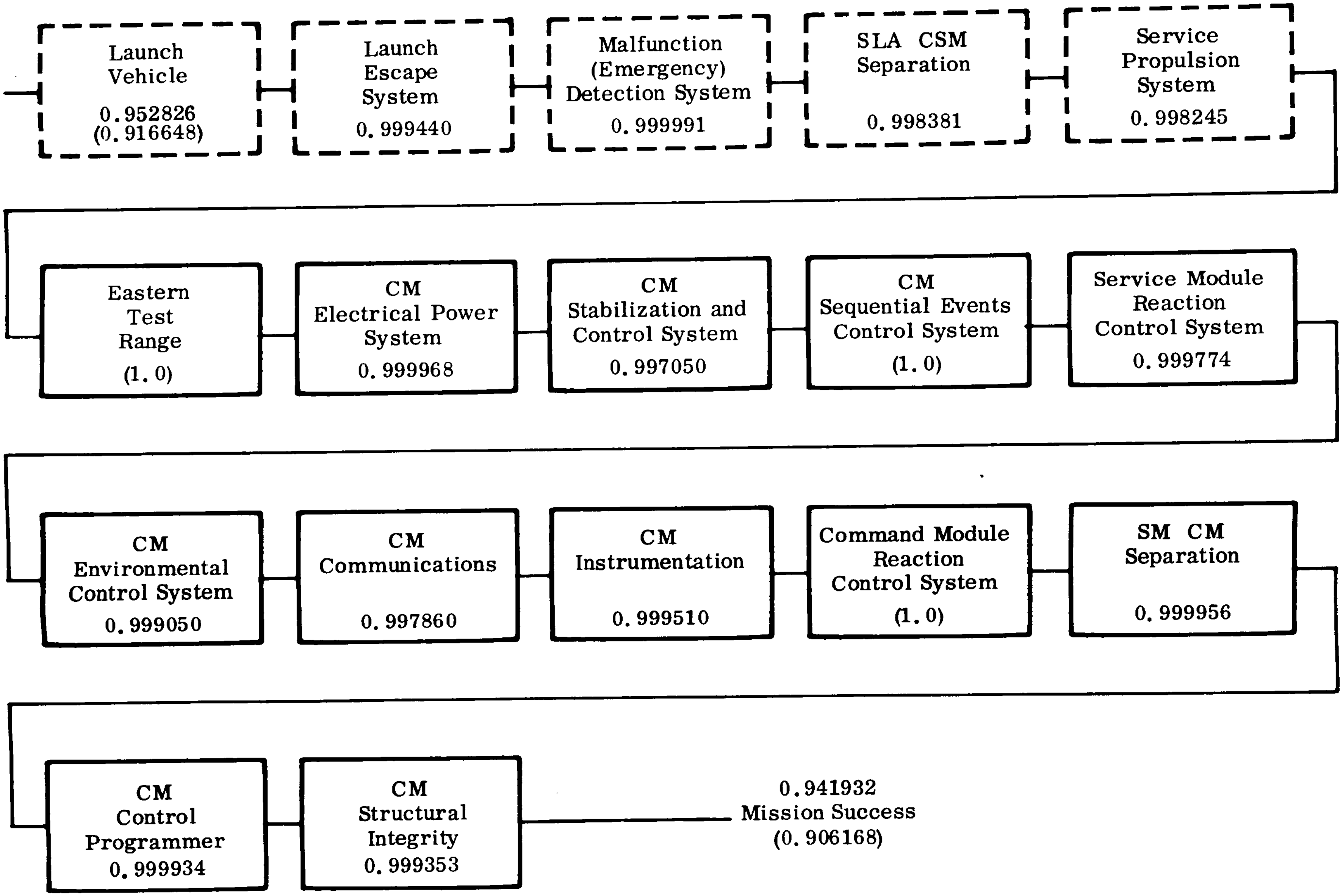


Figure A-3(l). Prediction Model for Mission Success, Beginning of SM CM Separation to Beginning of Entry 0.05g, Phase 13

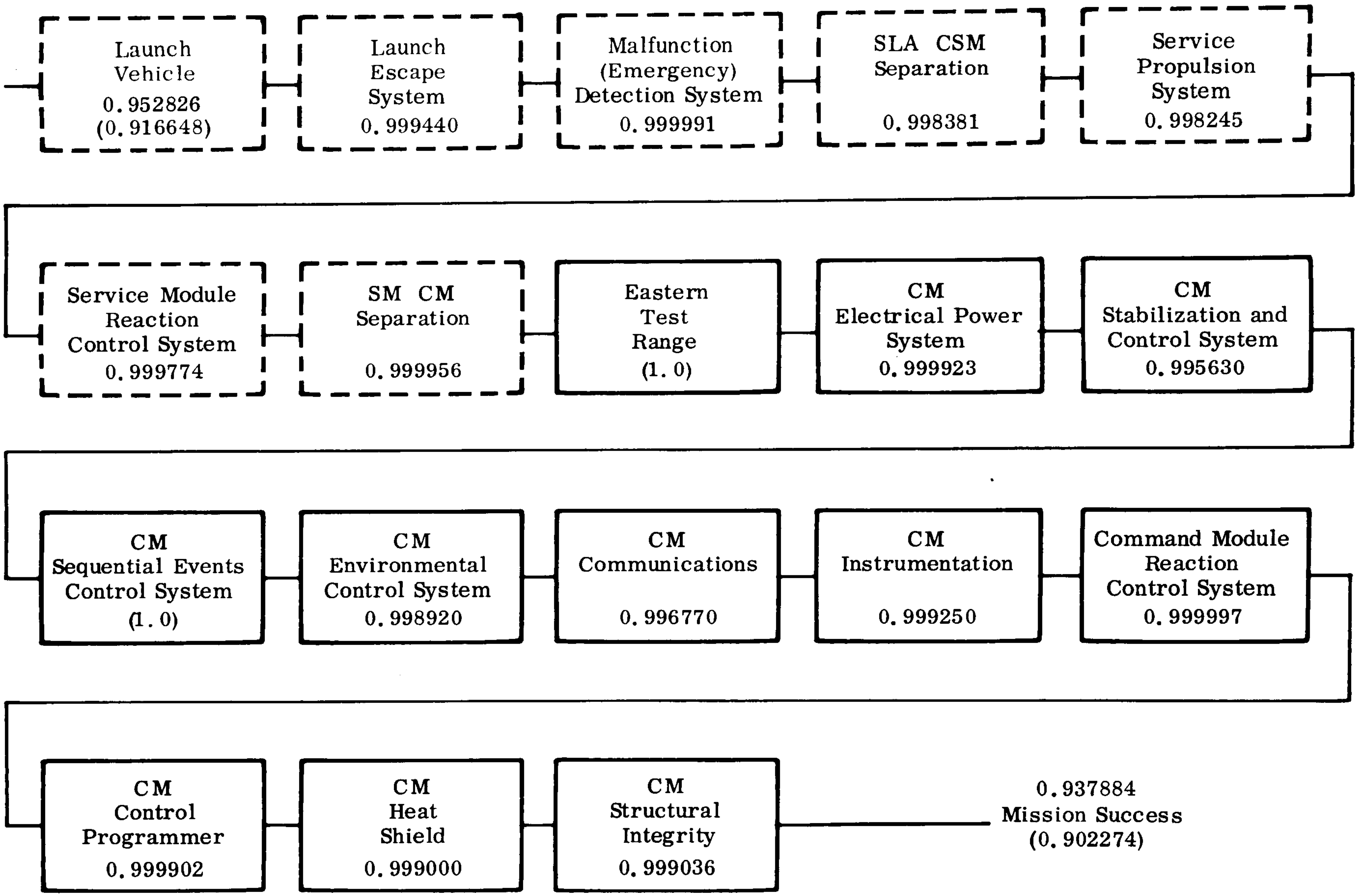


Figure A-3(m). Prediction Model for Mission Success, Entry 0.05g through Forward Heat Shield Jettisoning, Phase 14

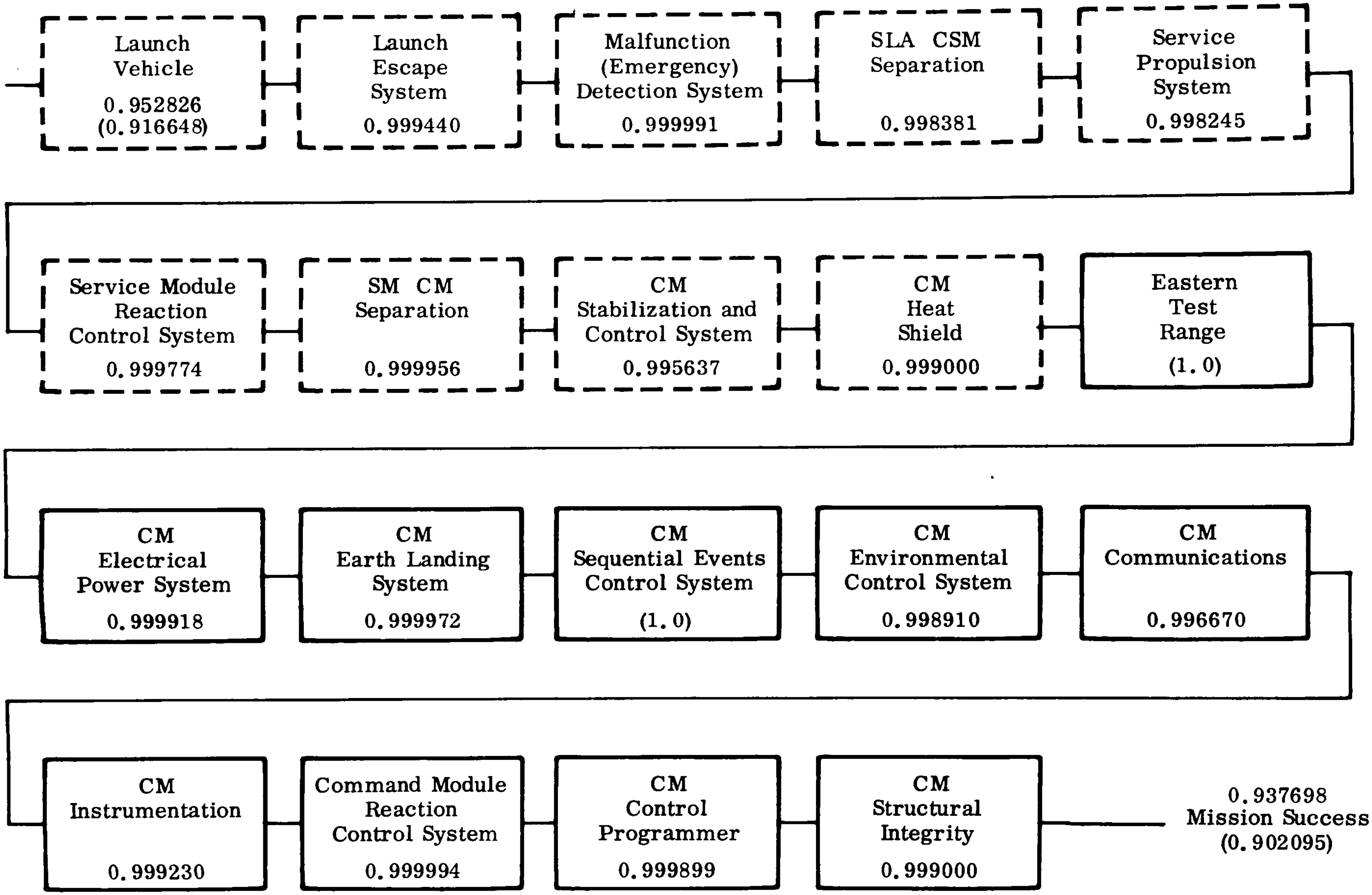


Figure A-3(n). Prediction Model for Mission Success, Forward Heat Shield Jettisoning to Touchdown, Phase 15

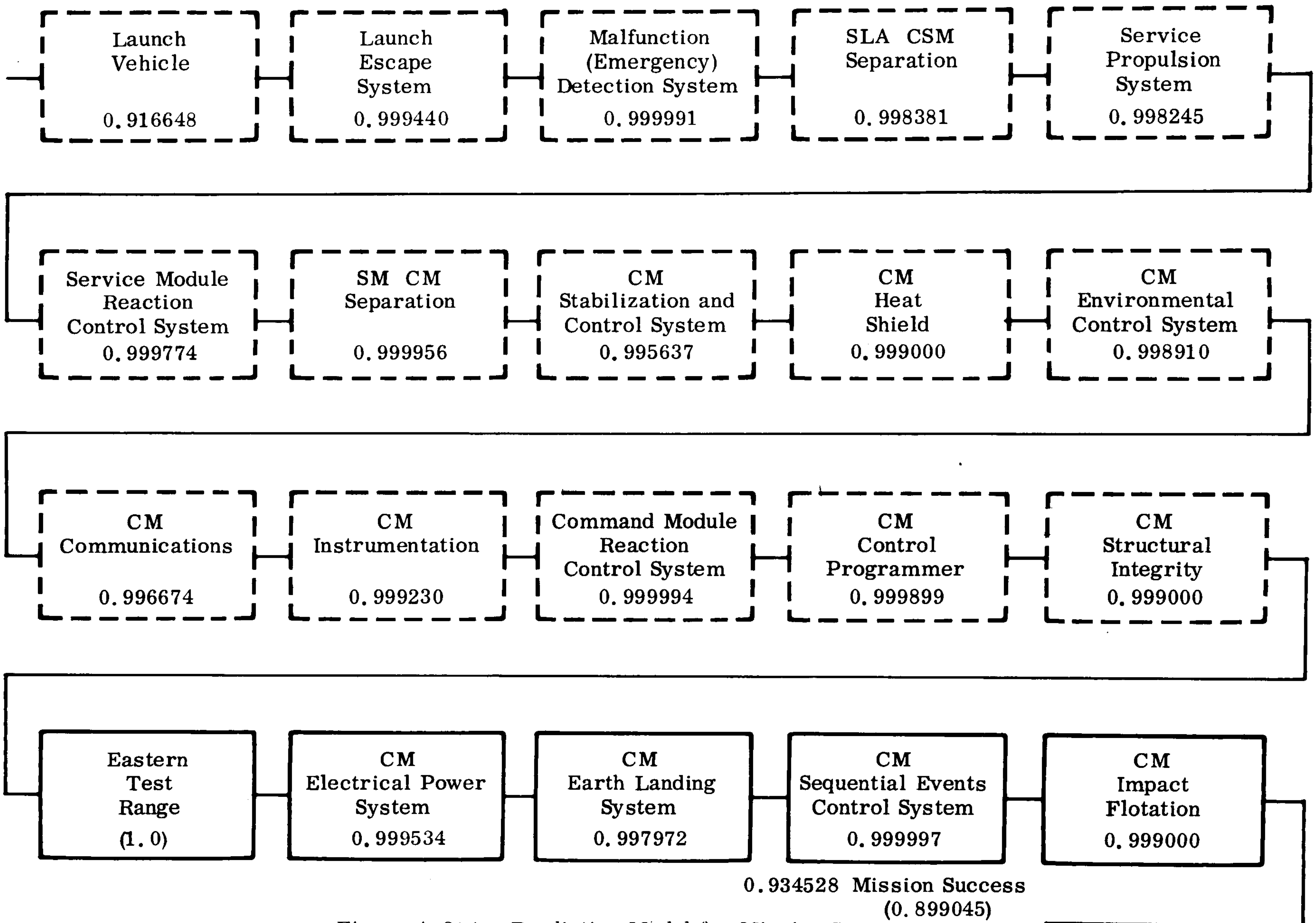
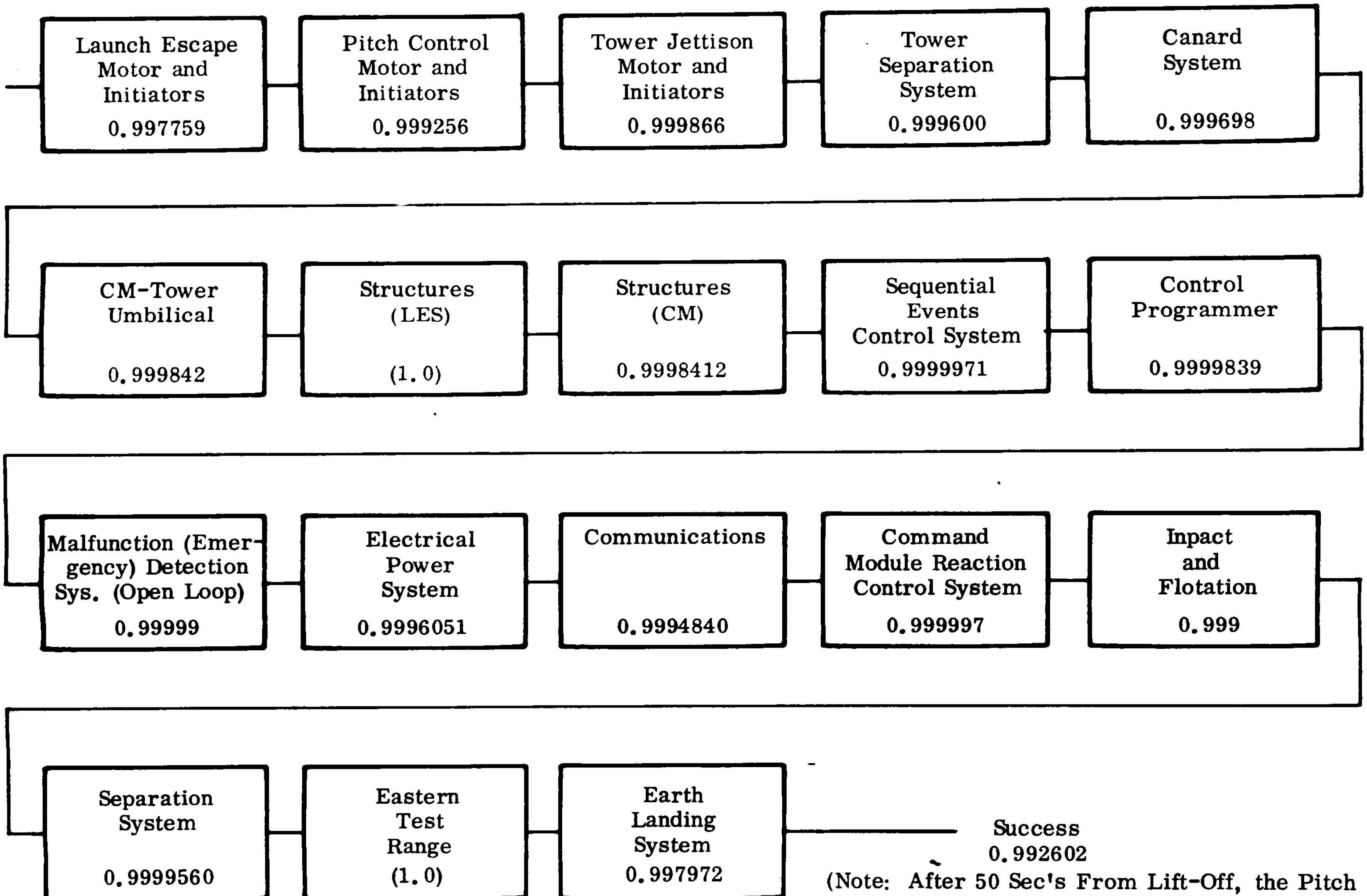


Figure A-3(o). Prediction Model for Mission Success, Touchdown to Retrieval, Phase 16



(Note: After 50 Sec's From Lift-Off, the Pitch Control Motor is not Needed.)

Figure A-4. Prediction Model for Launch Escape System Contingency (Sheet 1 of 2)

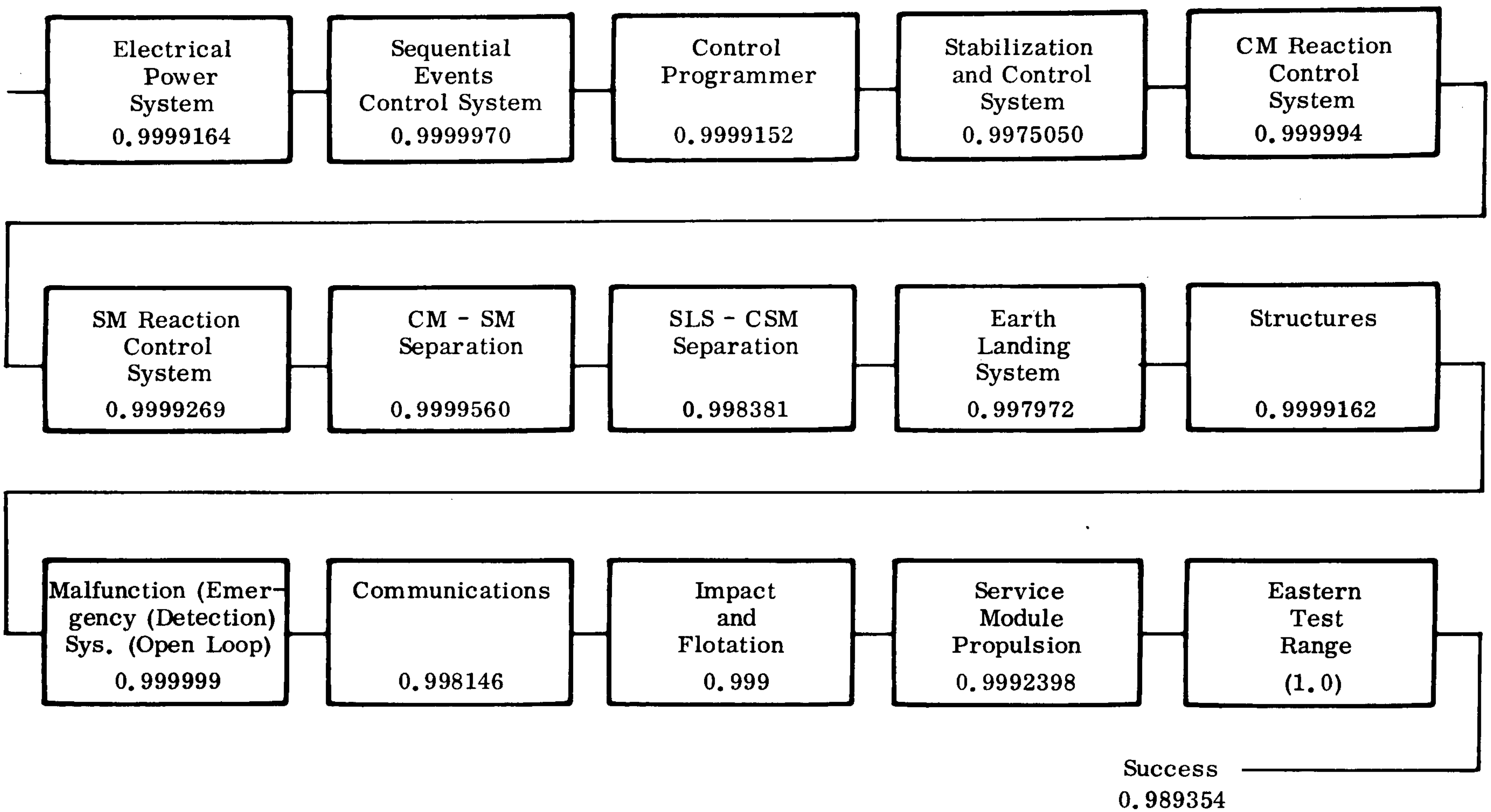
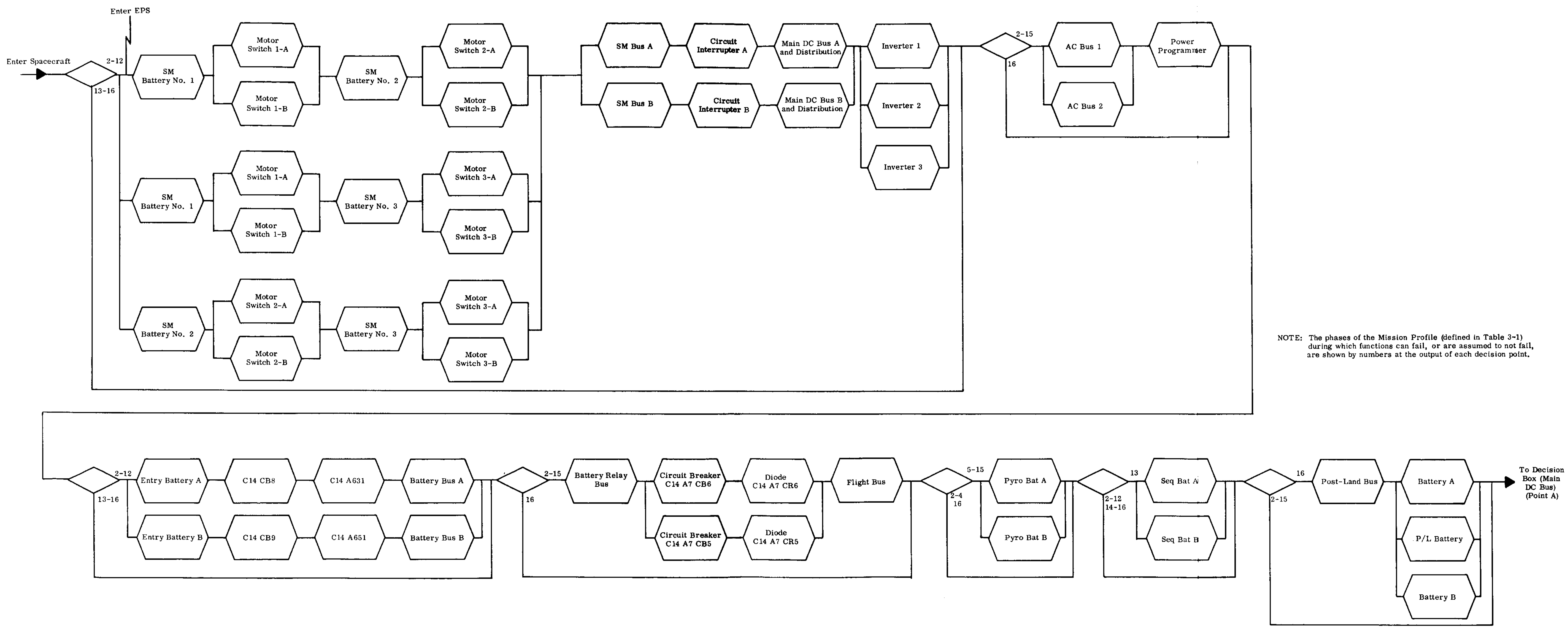
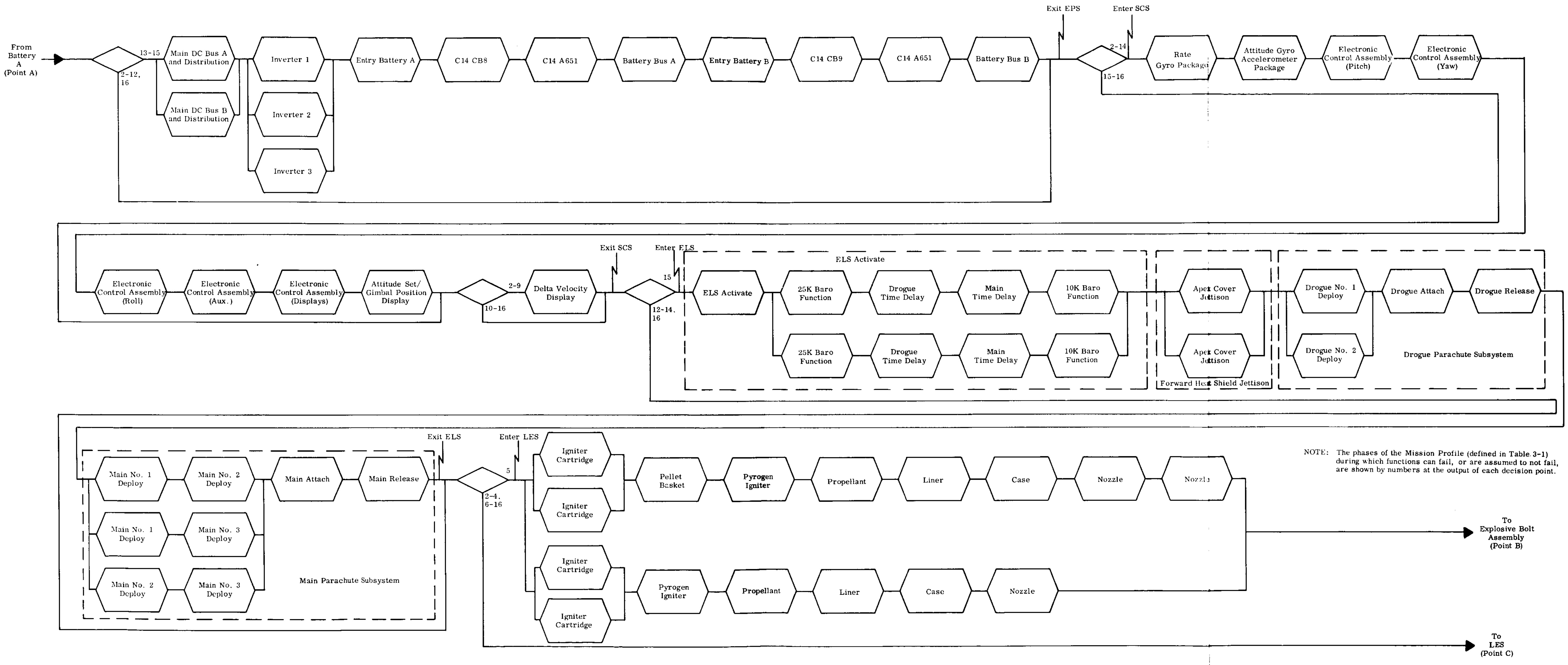


Figure A-4. Prediction Model for Service Propulsion System Contingency (Sheet 2 of 2)



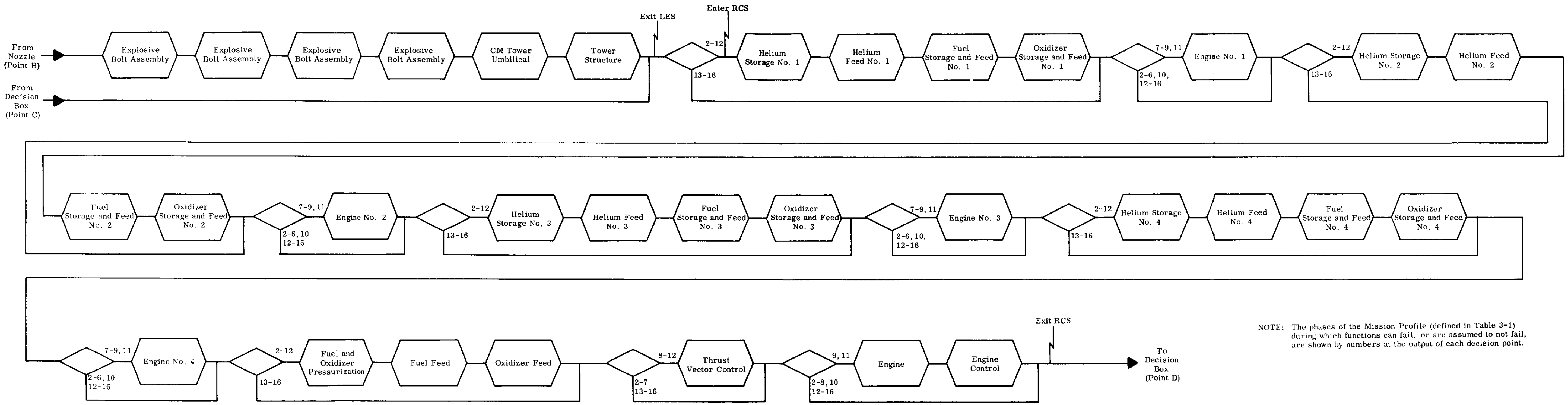
NOTE: The phases of the Mission Profile (defined in Table 3-1) during which functions can fail, or are assumed to not fail, are shown by numbers at the output of each decision point.

Figure A-5. Spacecraft 009 Mission Success Logic (Sheet 1 of 5)



NOTE: The phases of the Mission Profile (defined in Table 3-1) during which functions can fail, or are assumed to not fail, are shown by numbers at the output of each decision point.

Figure A-5. Spacecraft 009 Mission Success Logic (Sheet 2 of 5)



NOTE: The phases of the Mission Profile (defined in Table 3-1) during which functions can fail, or are assumed to not fail, are shown by numbers at the output of each decision point.

Figure A-5. Spacecraft 009 Mission Success Logic (Sheet 3 of 5)

[redacted]

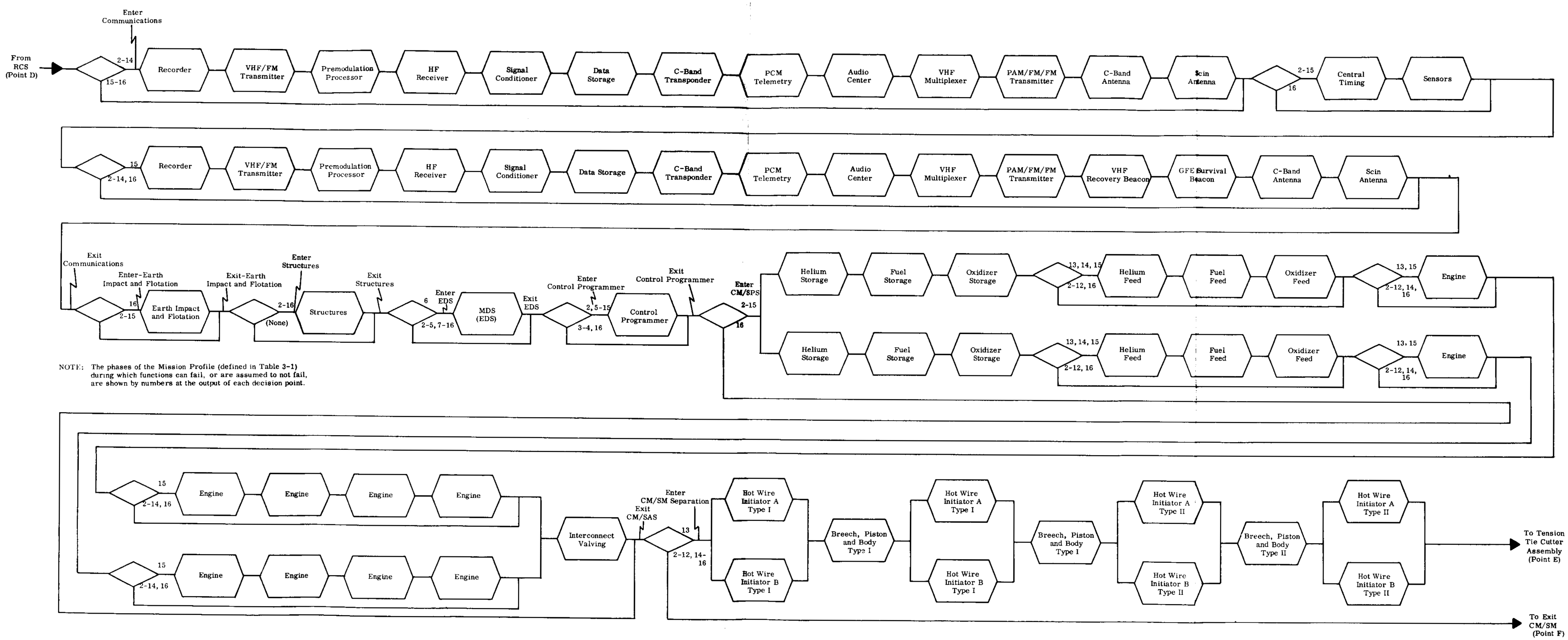
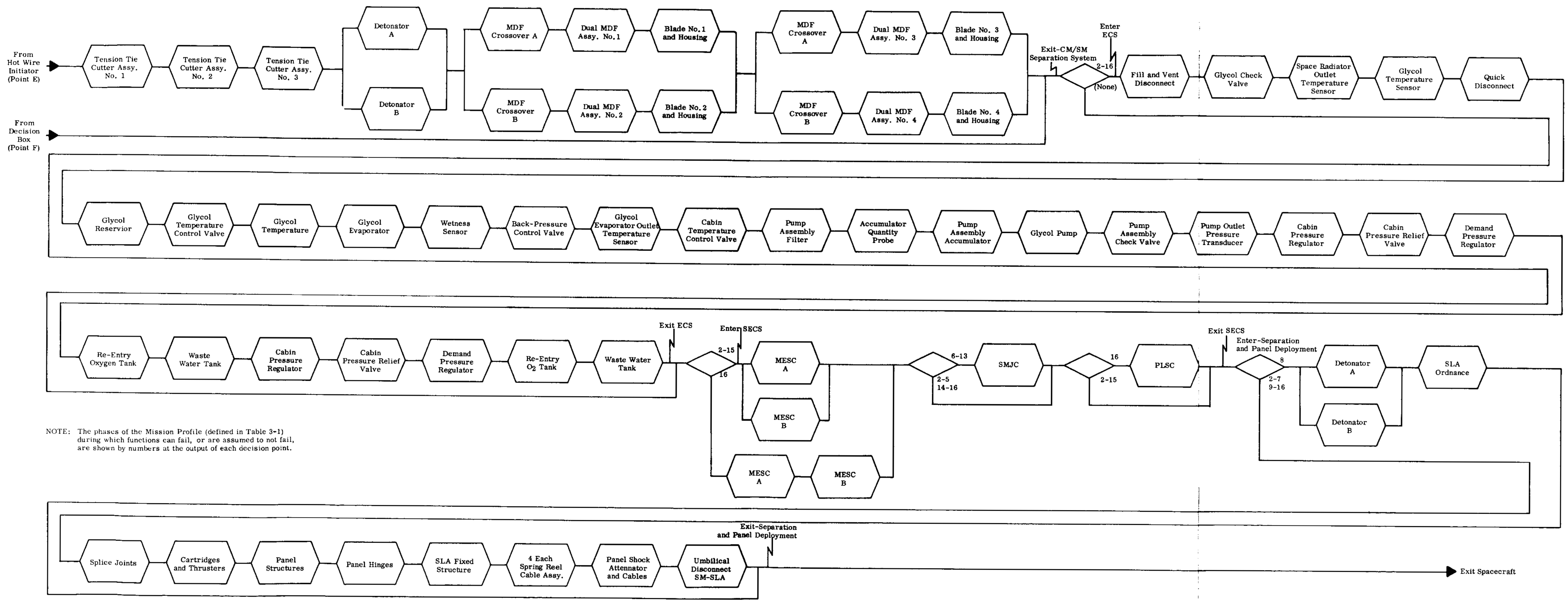


Figure A-5. Spacecraft 009 Mission Success Logic (Sheet 4 of 5)

[redacted]



NOTE: The phases of the Mission Profile (defined in Table 3-1) during which functions can fail, or are assumed to not fail, are shown by numbers at the output of each decision point.

Figure A-5. Spacecraft 009 Mission Success Logic (Sheet 5 of 5)

APPENDIX B

ACRONYMS AND ABBREVIATIONS

AR&QA	Apollo Reliability and Quality Assurance Office
AS	Apollo Saturn
ASPO	Apollo Spacecraft Program Office
CCSD	Chrysler Corporation Space Division
CM	Command Module
CSM	Command and Service Module
DAC	Douglas Aircraft Company
ECS	Environmental Control System
ELS	Earth Landing System
EPS	Electrical Power System
ETR	Eastern Test Range
FMECA	Failure Mode Effects and Criticality Analysis
FRR	Flight Readiness Review
GAEC	Grumman Aircraft Engineering Corporation
GE/ASD	General Electric Company, Apollo Support Department
IBM	International Business Machines Corporation
IU	Instrument Unit
KSC	Kennedy Space Center
LEM	Lunar Excursion Module
LES	Launch Escape System
MDS(EDS)	Malfunction (Emergency) Detection System
MSC	Manned Spacecraft Center
MSFC	Marshall Space Flight Center
NAA	North American Aviation, Inc.
NASA	National Aeronautics and Space Administration
OMSF	Office of Manned Space Flight
PFRR	Preflight Readiness Review
RCS	Reaction Control System
S/C	Spacecraft
SCS	Stabilization and Control System
SECS	Sequential Events Control System
S-IB	Saturn IB Stage of Launch Vehicle
S-IVB	Saturn IVB Stage of Launch Vehicle

APPENDIX B

ACRONYMS AND ABBREVIATIONS (Continued)

SLA	Spacecraft Launch Vehicle Adapter (or Spacecraft LEM Adapter)
SM	Service Module
Spacecraft 009	The Spacecraft Assigned to the 201 Mission
SPS	Service Propulsion System

APPENDIX C

LIST OF REFERENCED DOCUMENTS

1. NASA/OMSF, Apollo Program Flight Mission Directive for Apollo-Saturn 201 Mission, M-D-MA-2240. 061, 13 May 1965

2. NASA/OMSF, Apollo Program Specification (U), SE-005-001-1, May 1965, CONFIDENTIAL; and its Appendix AS-201, May 1965

3. NASA/OMSF, Apollo Reliability and Quality Assurance Program, Quarterly Status Report (U), RA018-001-1, Third Quarter 1965, 8 October 1965, CONFIDENTIAL

4. NASA/OMSF, Apollo Reliability and Quality Assurance Program, Quarterly Status Report (U), RA018-001-1, Second Quarter 1965, 9 July 1965, CONFIDENTIAL

5. NASA/OMSF, Apollo Reliability Analysis Review and Estimation Guidelines, RA006-007-1, November 1965

6. NASA/OMSF, Reliability Program Provisions for Space System Contractors, NPC 250-1, July 1963

7. NASA/MSC, Operational Support Plan for Apollo Mission SA-201, Flight Control Division, 27 May 1965

8. NASA/MSC, Program Apollo Flight Mission Directive for Mission A-201 (AFRM 009), NASA Program Apollo Working Paper No. 1153, 14 December 1964

9. NASA/MSC, Telegraphic Message to J. C. Cozad (NAA/S&ID) from C. L. Taylor (MSC/ASPO), dated 20 July 1965

10. NASA/MSC, Spacecraft Reference Trajectory SA-201, Internal Note No. 65-FM-14

11. NASA/MSC, Telegraphic Message, Mission AS-201 SPS abort Timeline for Setting Timing in Control Programmer Abort Timer, to R. Ridnour, RASPO NAA from C. L. Taylor MSC/ASPO, dated 3 November 1965

12. NASA/MSFC, Saturn IB Mission Plan and Technical Information Checklist, Volume 1, Revision 3, 1 April 1965

13. NASA/MSFC, Flight Mission Directive Apollo Saturn 201 Mission, 15 March 1965

14. NASA/MSFC, Saturn IB/SA-201 Flight Sequence, Dwg. 10M30152, 23 March 1965

15. NASA/MSFC, Procedure for Performing Systems Design Analysis, Dwg. No. 10M30111 Revision A, 26 June 1964

16. NASA Reliability Publication NPC 250-1, Reliability Program Provisions for Space System Contractors, July 1963

17. AVCO Reliability Engineering Data Series, Failure Rates, April 1962

18. Chrysler Corporation, Systems Design Analysis, Saturn S-IB-1 Stage, SDES 65-450, 9 August 1965

19. Chrysler Corporation, S-IB-1 Reliability Model, Summary of 10,000 Simulated S-IB-1 Flights, TN-RE-65-28, 2 April 1965

20. Collins Radio Company, Apportionment Study for the Apollo Communications and Data Support Subsystem, AR-162-1, 18 July 1962

21. Douglas Aircraft Company, Reliability Mathematical Model Saturn IB/S-IVB-201 Stage, Supplement 1, SM-46667, March 1965

22. Douglas Aircraft Company, Reliability Mathematical Model Saturn S-V/S-IVB Stage, SM-44748

23. GE/ASD, Apollo-Saturn 201 Mission Contingency Reliability Computations, ASD-MR-09-65-6, 17 September 1965

24. GE/ASD, Apollo-Saturn 201 Mission Success Computations (U), ASD-Mr-07-65-30, 16 July 1965

25. GE/ASD, Information for Reliability Prediction, ASD-5-05-64-1, 5 May 1964

26. Grumman Aircraft Engineering Corporation, Quarterly Reliability Status Report, LPR-550-2, 1 August 1963

27. International Business Machines Corporation, Preliminary Failure Mode, Failure Effect and Criticality Analysis for S-IU-201, 65-383-0034H, 23 May 1965

28. North American Aviation, Quarterly Reliability Status Report, SID-62-557-4

29. North American Aviation, Quarterly Reliability Status Report, SID-62-557-5

30. North American Aviation, Quarterly Reliability Status Report, SID-62-557-8

31. North American Aviation, Quarterly Reliability Status Report, SID-62-557-12

32. North American Aviation, Quarterly Reliability Status Report, SID-62-557-13

33. North American Aviation, Internal Letter T. A. Siciliano to L. B. Gray, AR-ISAA-65-50, 26 May 1965

34. North American Aviation, Minutes of Spacecraft Reliability Analysis Data Review Meeting, 30 June 1965 and 1 July 1965

35. North American Aviation, Sequential Events Control Subsystem, Mc, AP65-54, July 1965

36. North American Aviation, Schematic Diagram, Electrical Power System, Command and Service Module, Dwg. V14-945701